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THESIS

A SURVEY OF COMMERCIALLY AVAILABLE
EXPENDABLE HEAVY LIFT
LAUNCH VEHICLES

by

Joseph J. Mihal Jr.

June, 1994

Thesis Advisor: Dan C. Boger
Co-Advisor: Carl R. Jones

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A Survey of Commercially Available
Expendable Heavy Lift
Launch Vehicles

by

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B.S.M.E., Villanova University, 1986

Submitted in partial fulfillment
of the requirements for the degree of

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This thesis examines the launch vehicle options available to place a heavy payload, 10 tons, into a low-earth- orbit. The study provides the current status of the space launch vehicle market for heavy lift launchers. The following launchers are looked at in detail: Titan 3, Proton, Energia, Long March 3, Ariane 5, and the H-2. Vehicle design history and launch record are examined. Each launcher is then examined and broken down by stages, including the payload sections. A typical launch sequence is included for each vehicle. Finally, the cost of the various launchers is examined. Conclusions regarding the future need of heavy lift launch vehicles and a look at the current political environment is made.

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I. INTRODUCTION

The United States of America once was the undisputed leader in space vehicle launches in the Free World. If you wanted to put anything into space, you had to go to America. This all changed in 1982.

In 1982 the European consortium, Arianespace, began commercial launches. This ended the monopoly the United States had on commercial launches in the West.

The space shuttle was proving to be a reliable method of getting large payloads into space in 1983. The United States made a fateful decision that year; it was decided that all government payloads would be launched on the space shuttle.

In 1984 the Commercial Space Launch Act was passed. The aim of this Act was to commercialize the Expendable Launch Vehicle (ELV) industry and thus let United States manufacturers compete on the worldwide markets. The reality of the two decisions was that the commercial ELV industry was put on the brink of failure. The shuttle, being a subsidized launcher, was able to set prices which were so low that it was hard for a commercial venture to compete for launches. The requirement that all government launches use the shuttle quickly reduced the number of available payloads down to a small fraction of the original number. [Ref. 1:p. 7]

As a result of the two decisions, some launch vehicle contractors had to lay off workers and close down production lines due to the low demand. The major contractors affected in the United States were those who manufactured the Atlas and the Delta vehicles. Martin Marietta, manufacturer of Titan, was spared the initial shock when it was awarded a U.S. Air Force contract for ten Titan IVs to act as a backup for the shuttle.

Everything changed in January of 1986. The loss of the space shuttle Challenger and her crew marked the beginning of a new era for the ELV in the United States. Following the loss of the Challenger, the United States suffered a rash of launch vehicle failures. The Reagan Administration, in August of 1986, directed that the shuttle could no longer carry any commercial payloads, foreign or domestic, except for national security or foreign policy reasons, and that NASA was not allowed to provide any ELV services. [Ref. 1:p. 8] This was the saving grace for the U. S. commercial ELV industry.

The current United States ELVs are based on 25 to 40 year old technology. The infrastructure required to support these vehicles is deteriorating rapidly and is costly to maintain. [Ref. 2:p. 15] Gene Sevin, a space systems acquisition official in the Defense Department said, "Everything we do is terribly inefficient." [Ref. 3:p. 23] This comment was made regarding a comparison of the Arianespace launch support team in French Guiana, which is composed of approximately 900

people, to the 29,000 people required to support the Kennedy Space Center and Cape Canaveral Air Force Station in Florida.

The point is that there are other countries out there who are producing ELVs which can meet and or exceed the current United States ELV program in terms of lift capability and cost. In the area of heavy lift ELVs the nations of the United States, Russia/CIS, China and Japan all have vehicles which can place ten tons or more into a Low Earth Orbit (LEO). Arianespace is on the verge of launching its heavy lift vehicle. These are the vehicles which will be compared in this study. The vehicles include the Titan III, from the United States, the Proton and the Energia from Russia/CIS, Ariane 5 from the European Space Agency, the Long March 3 from China, and finally the H-2 from Japan.

II. TITAN

A. HISTORY

1. Development

The Titan family of launch vehicles can trace its beginnings back to the early days of the Cold War and the need for Intercontinental Ballistic Missiles (ICBMs). As with a majority of early United States space launch vehicles, including Atlas and Delta, the technology used to develop them originated from ballistic missile technology. Today's Commercial Titan represents the fifteenth variant for Martin Marietta, the company responsible for the production of Titan, on a design which was first conceived in 1955. Figure 1 shows the Titan family of launch vehicles.

Titan I became the nation's first two-stage Intercontinental Ballistic Missile. It was designed to use fuels which allowed it to maintain flight readiness over extended periods of time; this allowed it to be the United States' first silo-housed ICBM. Titan I provided a proving ground for many design and structural advances which were later incorporated into Titan II.

Titan II, the follow-on to Titan I, started development in 1962 and eventually became the United States' largest land-based missile during the early 1960's. Titan II

used a different fuel combination than its predecessor; it used A-50 hydrazine and N204 nitrogen tetroxide. This combination is still used today. [Ref. 4:p. 105] The Martin vehicle was also modified and served as a man-rated launch vehicle for NASA's Gemini program. There were twelve successful launches of the Gemini capsule on board Titan II boosters. The Titan II is making a comeback as a space booster. The old ICBMs are being converted to launch U.S. Air Force payloads into orbit. [Ref. 5:p. 264]

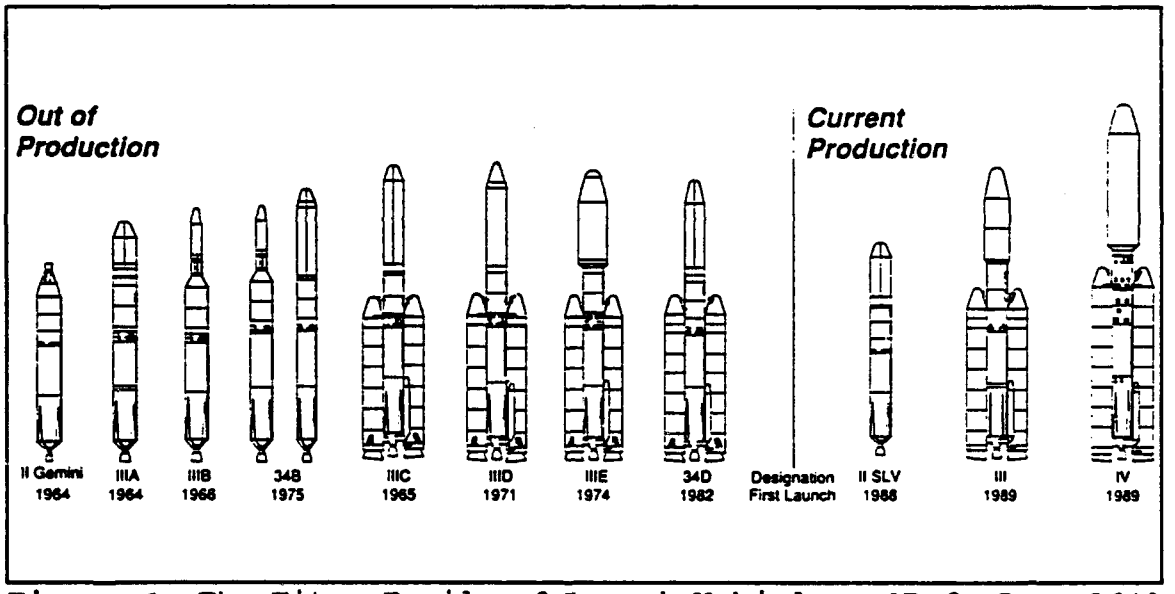


Figure 1. The Titan Family of Launch Vehicles. [Ref. 5:p. 264]

The next member of the Titan family, Titan III, began its development at the end of 1962. The concept of the Titan III design was for the construction of a modular vehicle which would satisfy a wide variety of the then-projected heavy lift requirements. The system was centered around a redesigned Titan II ICBM, which required improved guidance systems and

structural strengthening. In addition, strap-on solid-propellant boosters would also be utilized, plus the addition of a fourth stage which could maneuver while in orbit (a transtage). [Ref. 6:p. 123] One of the primary missions which drove the Titan III design was the U. S. Air Force's own manned space program, which included a reusable delta-winged glider called the Dyna Soar.

The Titan III family has had the largest number of variants based on the original Titan ICBM. The Titan IIIA was completed in the middle of 1964 and consisted solely of the core vehicle. The IIIB was very similar to the IIIA except that it did not use the IIIA's transtage or the inertial navigation systems. Instead it was designed to use an Agena upper stage. The IIIC, which was operational only a year and a half later, consisted of the core vehicle plus strap-on boosters The Titan IIID entered service in 1971. The IIID's major difference from the IIIC was the absence of the transtage. The IIIE, which entered service in 1974, was a NASA managed program vice a U. S. Air Force managed one. It was designed and used to launch planetary probes. The IIIE was a IIID redesigned to carry a General Dynamics Centaur upper stage. The next variant of the Titan III family, the Titan 34D, was designed to provide a launch capability that filled the gap between then existing launch vehicles and the Space Shuttle. The first 34D was successfully launched in October of 1982.

The current Titan stable of launch vehicles is comprised of the Titan II Space Launch Vehicle (SLV), the Titan III, and the Titan IV. These launch vehicles are the direct result of the various Titans described earlier.

The Titan IV, was developed as a result of the problems the shuttle program was having in the early 1980's. The U.S. Air Force, in 1985, contracted with Martin Marietta for a modified Titan 34D, which was originally called a Complementary Expendable Launch Vehicle (CELV). The modifications included using a seven-segmented strap-on booster design, stretched core stages and a 16.7 ft diameter payload fairing. This vehicle became known as the Titan IV. The Titan IV is currently the main method for United States Department of Defense to launching heavy payloads.

The Titan II SLV was begun at the same time the Titan IV project was started. The Titan II SLV project was to save money by using the Titan II ICBMs, which had been taken out of active service, refurbishing them and utilizing them as space launch vehicles. There are 55 Titan II ICBMs available for use, of which the U. S. Air Force has currently ordered 14 such vehicles to be refurbished. The first successful launch of a Titan II SLV occurred on 5 September 1988. [Ref. 5:p. 265]

In 1986 Martin Marietta announced the availability from the spring of 1989 of a commercial version of its Titan 34D, called the Commercial Titan, or Titan III. This decision

can be linked to the Challenger accident in 1986, after which time the U. S. Government decided to offload commercial payloads from the Space Shuttle. Titan III is a 34D with a stretched second stage and new fairing, which is 13.1 feet in diameter. The new fairing allows for single or dual shuttle-class payloads with the addition of perigee kick motors. The first Titan III was successfully launched 1 January 1990 carrying two communication satellites. [Ref. 7:p. 315]

The Titan family of launch vehicles has come a long way since its early development as ICBMs in the 1950's. The Titan has come full circle, with a variant of one of its earliest ICBMs being refurbished and used as a current space launch vehicle. The current Titan IV represents the greatest lift-capacity expendable launch vehicle available from a U. S. manufacturer.

2. Success Rate

As a family of launch vehicles, Titan has proven to be a very reliable method of placing objects into space. The overall success rate for the entire Titan family is 92.7%. This is based on data through the middle of 1993. Out of a total of 178 vehicle launches, only 13 were failures. Figure 2 gives a breakdown by year of the Titan launch history. [Ref. 8:p. 223]

The Titan II series of boosters maintains a 100% reliability rating, having been used successfully in 15 of 15

launch attempts. The Titan II booster's first successful launch occurred on 8 April 1964, with the launching of the manned Gemini 1 capsule. The Gemini series of launches continued through Gemini 12, which was launched on 11 November 1965. The new Titan II program, discussed above, had it first successful launch on 5 September 1988. The program continues through the present day.

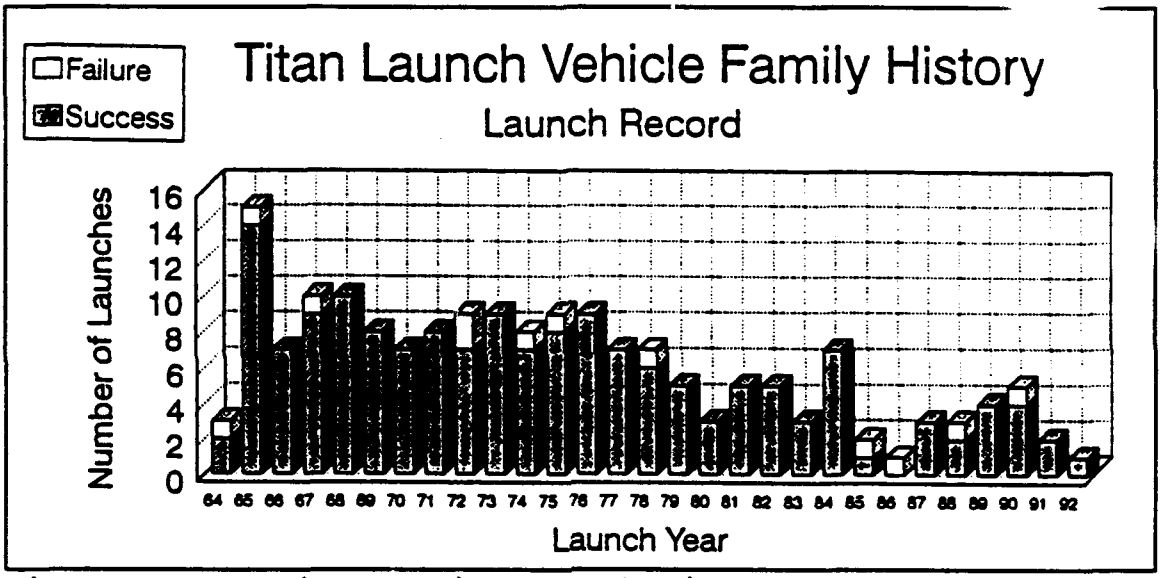


Figure 2. The Titan Family Launch History.

The Titan III family of launchers has had the largest use and also has had the most failures. Overall the Titan III success rate is 144 successful launches out of 156 attempts. This provides a 92.3 success rate for that family of boosters. The Titan IIIA has a 75% success rate with three successful launches out of four attempts. The Titan IIIB has a 96.3% success rate with 54 successful launches out of 54 attempts. The Titan IIIC has a 92% success rate with three failures out

of 36 attempts. The Titan IIID has a flawless record of 22 launches out of 22 attempts. The Titan IIIE has a 85.7% success rate with six successful launches in seven attempts. Titan 34B has a success rate of 92.9% with 13 successful launches out of 14 attempts. The Titan 34D has an 80% success rate with three failures in 15 attempts. Finally Titan III, Commercial Titan, has a 67% success rate with two successful launches out of three attempts.[Ref. 8:p. 223]

The failures which have occurred have been spread throughout the entire family of Titan III. No clear link can be established between failures, and therefore no fundamental design flaws were found. It is interesting to note that out of the 12 failures from 1964 through 1993, five of them have been related to the transtage. The transtage suffered a pressurization system failure on the first Titan IIIA flight in 1964. Then, in 1965, again the transtage failed when the engine did not shut down, this time as part of a Titan IIIC. In 1960, aboard a Titan IIIC, the transtage guidance system failed. The same type of failure occurred again in 1975 aboard a Titan IIIC. In 1988, a transtage pressurization system failure occurred as part of a Titan 34D flight.[Ref. 5:p. 266]

The remaining seven failures have had various causes. The 1965 Titan IIIC failure was attributed to attitude thrusters sticking open. The 26 August 1966 failure of a Titan IIIC was the result of the payload fairing failing. In

1967 a Titan IIIB failed because the second stage engine lost thrust. A hydraulic pump failure in the second stage of a Titan IIIC was the reason for failure of a 25 March 1978 launch. A Titan 34D failed on 28 August 1985 when a first stage propellant feed system failed. In 1986, the same year as the Challenger accident, a Titan 34D failed as a result of a solid motor thermal insulation failure. On 14 March 1990 a Commercial Titan failed due to miswiring of the payload separation system. [Ref. 5:p. 266]

The Titan IV success rate is 85.7% with six successful launches out of seven attempts. The failure of a Titan IV on 2 August 1993, is suspected to be the result of a failure in one of the solid rocket motors. The suspected segment is the same one which was discovered to have 60 individual flaws, during an inspection at Cape Canaveral Air Force Station, and was sent back to the manufacturer for repairs. The problems included debonding between the casing and its liner and multiple cavities on the interior of the propellant. [Ref. 9:p. 1] The Titan IV is currently only used to launch DoD payloads.

It should be noted that the figures providing the family launch history of the various launchers uses data that strictly concerns whether a launch was successful or not. The information provided in these figures is not designed to show reliability of a particular launcher. As launchers are

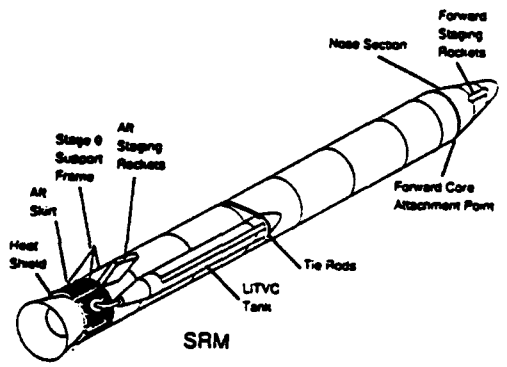
developed, technological improvements increase the final products reliability.

B. TITAN III VEHICLE DESCRIPTION

The Commercial Titan is composed of a core element, consisting of two stages and two Solid Rocket Motors (SRMs), which provide the initial lift-off thrust.

1. Stage 0 Solid Rocket Motors

The SRMs are manufactured by United Technologies' Chemical Systems Division, San Jose, California. Each motor is comprised of five and one half segments, which are held together by 237 hand-placed clevis pins. [Ref. 5:p. 234] The SRMs provide all of the flight control and thrust for the initial phase of launch. Figure 3 shows the SRM.



Dimension:	SRM
Length	90.4 ft (27.6 m)
Diameter	10.2 ft (3.11 m)
Mass: (each)	
Propellant Mass	463K lb (210K kg)
Gross Mass	543K lb (247K kg)
Structure:	
Type	Monocoque
Case Material	Steel
Propulsion:	
Propellant	84% PBAN
Avg. Thrust (each)	1.4M lb (6.2M N) vac
Number of Motors	2
Number of Segments	5-1/2
Isp	271.6 sec vac
Chamber Pressure	934 psia (64.4 bar)
Expansion Ratio	8:1
Control-Pitch,Yaw,Roll	N2O4 Liquid Injection (effective ±5°)

Figure 3. The Titan Solid Rocket Motor. [Ref. 5:p. 269]

Aerodynamic stability of the booster is provided by canting the nozzle of the SRM six degrees from center line. Flight control is accomplished by using thrust vectoring. Thrust vectoring is provided by injecting nitrogen tetroxide through 24 flow injector valves, in four banks of six valves, which surround the exit nozzle. The hydraulically actuated injector valves control the flow of nitrogen tetroxide by changing the orifice opening which controls the flow of the fluid. Once the nitrogen tetroxide is injected it creates an oblique shock wave which changes the direction of the motor's exhaust gases and thereby allows control of the Titan III. A seven degree deflection can be obtained by opening all the valves in one bank. [Ref. 6:p. 125] The SRMs are attached to the core vehicle at the first stage at eight hardpoints which are located opposite one another along the yaw axis.

Each SRM is 90.4 feet in length and 10.2 feet in diameter. The propellant mass is 463,000 lb and each unit weighs 543,000 lb. The propellant used is 84% Polybutadiene Acrylonitrile Acrylic Acid (PBAN) which provides an average thrust of 1,400,000 lb vac with an Isp of 271.6 seconds vac. The nominal burn time for the SRM is 116 seconds and these motors are designed to burn until all the fuel is used; there is no shut down capability.

2. Stage 1

The avionics on board the launch vehicle sense the deceleration of the vehicle which is associated with SRM burnout and sends a command for the Titan core stages to take over. This is accomplished by igniting the first stage engines and jettisoning the SRMs. To prevent confusion with stage naming, the original Titan ICBM stage names were kept, and the SRMs became Stage 0. [Ref. 6:p.126]

The first stage is manufactured by Martin Marietta, Denver, Colorado. It is composed of two LR-87-AJ-11 engines (developed by Aerojet Propulsion Division), fuel and oxidizer tanks, intertank structure, required piping and a fore and aft skirt. Figure 4 shows the Titan first stage.

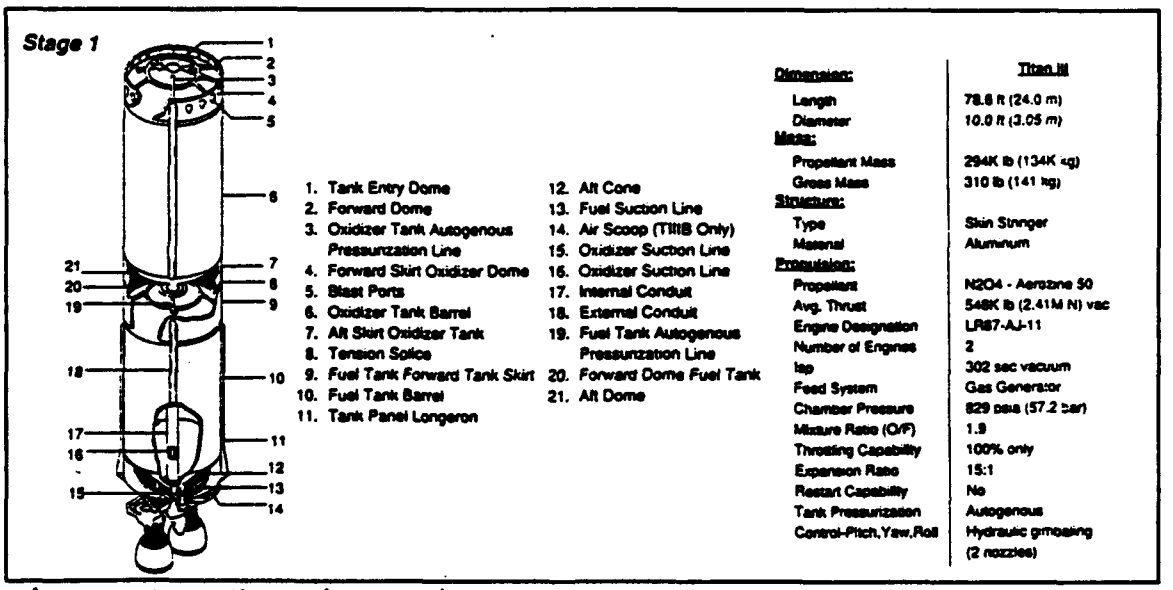


Figure 4. The Titan First Stage. [Ref. 5:p. 270]

The Model 11 Aerojet engines were first brought into service in 1968. Along with the second stage engine, the LR-

91-AJ-11, the LR-87-AJ-11 are the only U. S. engines which use a storable liquid oxygen/RP. [Ref 7:p. 342] This feature provides the Titan III the capability to meet critical launch windows without the special handling that is required for cryogenic fuels. The engine burns N204 - Aerozine 50, and provides an average thrust of 548,000 lb vac, with an Isp of 302 sec vac. There is no throttling capability with these engines, but there is a command shut down capability. These are burn-to-depletion engines.

Stage 1 is 78.6 feet tall and 10.0 feet in diameter. It has a total weight of 269,000 lb with a fuel load of 260,000 lb. It is built primarily of aluminum and uses both monocoque and semi-monocoque construction.

The fuel and oxidizer tanks are both constructed in the same manner. Each one consists of a dome on either end welded to a barrel-shaped center region. The tanks are structurally independent, which is done to prevent any mixing of the fuel and oxidizer in the event of a leak. The oxidizer tank is placed above the fuel tank and has a conduit to carry the oxidizer through the fuel tank to the engine. [Ref. 5:p. 270]

Stage 1 engine components are shielded from SRM exhaust by the use of an aluminum boattail heat shield. It protects those parts of the engine above the thrust chamber assembly above the throat. [Ref. 5:p. 270]

Pitch, roll and yaw control for the first stage is provided through hydraulic gimbaling of the thrust chamber. The driving force for the hydraulics is provided by the engine turbopump. Electronic signals from the guidance and flight control systems provide control to the turbopump.

3. Stage 2

The second stage is also produced by Martin Marietta of Denver, Colorado. It is comprised of the Aerojet LR-91-AJ-11 engine, oxidizer tank, fuel tank, intertank structure, transition assembly and an aft skirt. Figure 5 shows the Titan second stage.

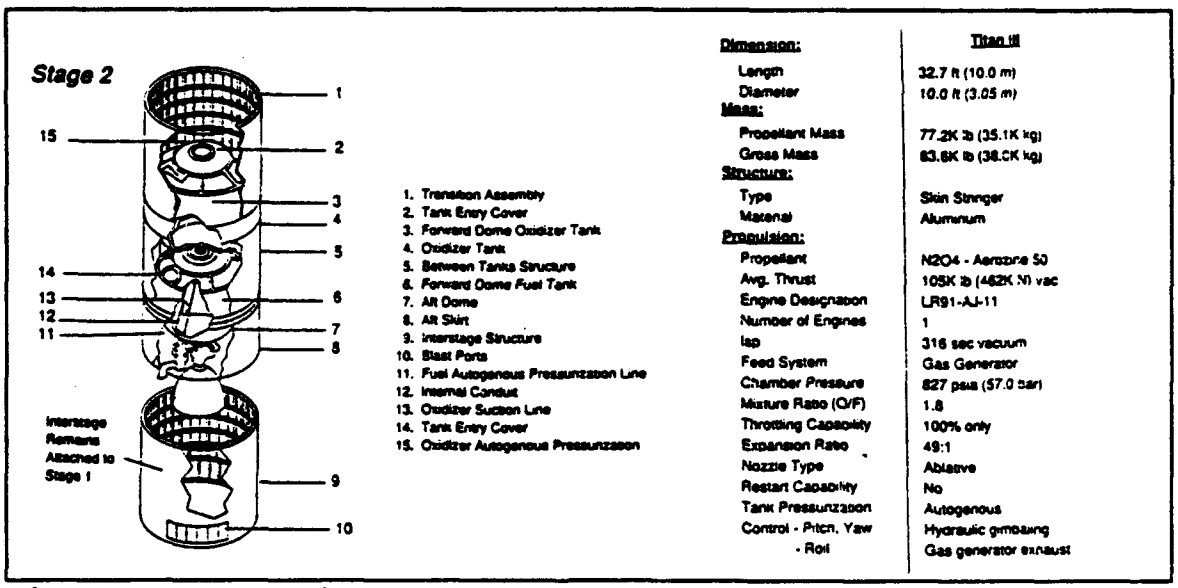


Figure 5. The Titan Second Stage. [Ref. 5:p. 271]

The second stage is 10.0 feet in diameter and stands 32.7 feet tall. It weighs 83,600 lb of which 77,200 lb is

propellant. It is constructed using the same techniques as the first stage and is also made of aluminum.

The LR-91-AJ-11 engine was first successfully flown in 1968, and has flown successfully on every Titan III mission since. This engine burns the same fuel as the LR-87-AJ-11 first stage engine. It is capable of producing 105,000 lb vac of thrust with an Isp of 316 seconds vac. There is no capability to adjust the output of the engine, i.e., no throttling capability.

The fuel and oxidizer tanks are spherical and are designed to hold 3,760 gallons of fuel and 4,200 gallons of oxidizer.[Ref. 4:p. 109] This quantity of fuel allows for a second stage burn time of approximately 225 seconds. The transition assembly is designed for use in coupling an upper stage or payload to the core Titan III vehicle.

Pitch and yaw control are provided for in the same manner as in the first stage, by hydraulic gimbaling of the thrust chamber. Roll control is accomplished through directing pump turbine exhaust through a swiveled nozzle to produce thrust.

Stage separation occurs once the onboard avionics sense vehicle deceleration, after burnout of the first stage. As the vehicle senses deceleration, the second stage is given the command to ignite while it is still attached to the first stage. The pressure developed from the second stage ignition is vented through blast ports located in the Interstage

structure, which connects the first and second stages, until the pyros ignite and separate the two stages. [Ref. 5:p. 271] After stage separation the Interstage structure remains with the first stage.

C. PAYLOAD SECTION

The Titan III has two payload configurations available. They are single and dual payload carrier options. The dual payload carrier is 53.5 feet in length and has a diameter of 13.1 feet and weighs 6,325 lb. The single payload carrier is 42.6 feet in length, 13.1 feet in diameter and weighs 4,990 lb. Figure 6 shows the dual payload carrier section.

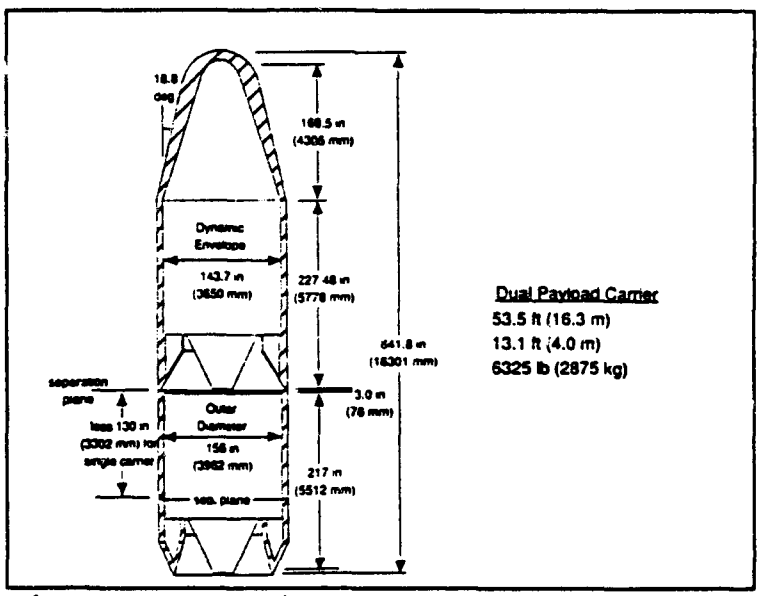


Figure 6. The Titan Dual Payload Carrier. [Ref. 5:p. 272]

The carrier is designed to hold the customer's spacecraft and attendant support systems; it also provides protection

during ground handling and the initial phase of flight through the atmosphere.

Both carriers are composed of a shroud and a payload extension module. These two items are produced by European manufacturers. Contraves of Switzerland produces the shroud, which is very similar in design to the shroud used in the Ariane IV program. Dornier of Germany produces the payload extension module. [Ref. 6:p. 143]

The shroud is made from carbon-fibre-epoxy aluminum honeycomb. Each payload has its own two-piece shroud. Hence, for the dual payload configuration there would be four shroud pieces. The shroud is topped by an aluminum nose cone.

The extension module is constructed using the same materials as the shroud. It is comprised of two sections, an aft section, which is used on all launches, and a forward skirt which is used in the dual carrier configuration, when it encloses the lower payload. [Ref. 6:p 144]

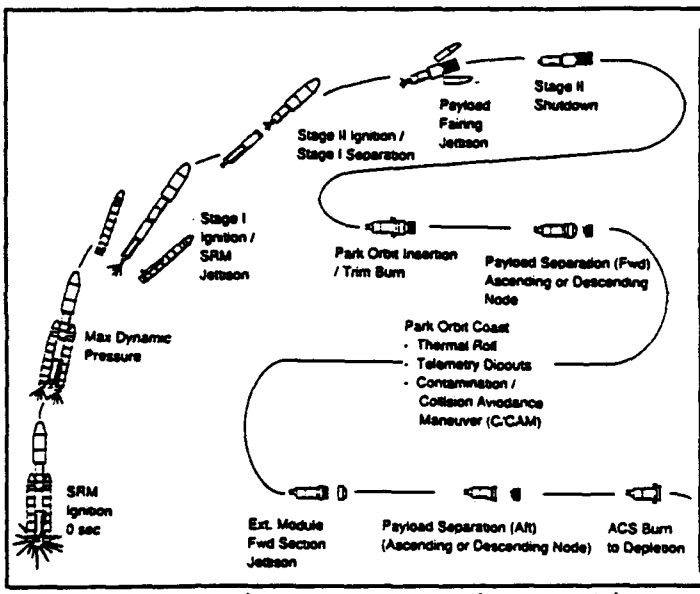
The payload section was designed to be compatible with numerous upper stages, depending on the customer's needs. The upper stages supported include the Martin Marietta Transtage, the McDonnell Douglas PAM-DII (Payload Assist Module), and OSC/Martin Marietta TOS (Transfer Orbit Stage). If neither of these systems are required the payload section also has the capability of providing two alternate methods of payload deployment. These systems are a spring ejection system, which

simply pushes the payload out, or a spin table that can rotate the payload to 70 rpm, prior to release. [Ref. 4:p. 111]

Precise orbit control of this section for the deployment of spun and spring ejected payloads is accomplished through the use of 12 Rocket Research MR-107 hydrazine motors. These motors produce 80-133 Newtons of thrust and are mounted in groups of four on the boattail. [Ref. 6:p. 143]

D. TYPICAL MISSION PROFILE

Currently Titan III is being launched from Cape Canaveral, Florida. The typical mission time, from launch until launch vehicle deorbit, is about three and one half hours, for the deployment of a dual satellite payload. Figure 7 shows a typical Titan III flight sequence.



Time (min:sec)	Events
00:00	Stage 0 Ignition
00:54	Max Dynamic Pressure
01:48	Stage 1 Ignition
01:56	Stage 0 Jettison
04:29	Stage 2 Ignition
04:30	Stage 1 Separation
04:40	Payload Fairing Separation
08:14	Stage 2 Shutdown
08:30	Park Orbit Insertion
08:40	Trim Burn
67:16	Deploy Payload 1
155:50	Deploy Payload 2
201:50	Launch Vehicle Burn to Depletion (Deorbit)

Figure 7. Titan III Typical Flight Sequence. [Ref. 5.p. 278]

The payload can be accessed directly until the T-10 day point. Access to the payload is provided through access panels until the T-3 hour point. At the T-2 hour point the mobile service tower is moved to its parked position. The countdown can proceed all the way to the T-5 minute point and have a countdown hold initiated. After the T-5 point countdown recycling will occur. At 00:00 the SRMs of stage 0 are ignited.

The Titan III launch vehicle, which had been resting on the SRM aft shroud, lifts off the launch pad powered strictly by the SRMs. Once the tower is cleared, the guidance system, located in the second stage, commands a pre-programmed pitch and roll maneuver to place the vehicle in the proper ascent profile. This maneuver is performed using the system of nitrogen tetroxide injection which was described earlier.

At time 00:54, the maximum dynamic pressure is reached at an altitude of 36,000 feet. As the SRMs reach the end of their burns, Stage 1 ignites at 01:48, followed by Stage 0 separation at 01:56. At this time the core vehicle is approximately 29 miles in altitude and accelerating at 4,170 mph. [Ref. 4:p. 112] Stage 1 burns for approximately 164 seconds; stage 2 is ignited at time 04:29 followed immediately by stage 1 separation at 04:30. This occurs at an altitude of approximately 72 miles. Ten seconds later, at time 04:40, the payload fairing is jettisoned. [Ref. 4:p. 112]

Stage 2 will burn for a nominal time of 225 seconds. Shutdown occurs at about 08:14. The vehicle and its payload enter into a parking orbit, utilizing attitude control thrusters, at 08:30. From this orbit, depending on payload missions and orbits, the payloads are either placed into orbit or the vehicle and payload is oriented so that the transtage can take the payload to the proper orbital plane. At approximately 201:50 the launch vehicle deorbits, concluding the mission.

E. COST

When the Commercial Titan was first announced, a company seeking to launch a payload had the option of buying the whole payload section or part of a dual payload section. The latter was the case for the first launch of the Titan III, when a British and a Japanese satellite were launched. In June of 1989 Martin Marietta announced that "half rides" were no longer being provided. It had decided that it would only sell dedicated rides. The customer who purchased the vehicle had the option of launching either one or two payloads.

The estimated launch price for the Titan III, based on 1991 dollars was between 130-150 millon dollars. This price did not include an upper stage, if required. This is the base price for the Titan III and the dual payload launch shroud. [Ref. 5:p. 268]

III. PROTON

A. HISTORY

CIS launch vehicles are referenced by three different names. There is the Russian designation, the United States designation and the Sheldon designation. The Russian designation results from the practice of naming the booster after its first payload; examples of this are the Kosmos and the Proton. The United States designations come from the Department of Defense which assigns a numerical designation to each new vehicle based on order of appearance. Examples of this method are, for the Proton, SL-9 launched in 1965, SL-12 launched in 1967 and the SL-13 first launched in 1968. The Sheldon names are based on indicators for specific families of launch vehicles. Dr. Charles Sheldon of the U. S. Library of Congress is most commonly associated as the founder of this method. The Proton family of vehicles has the Sheldon designator D.

1. Development

The Proton launch vehicle, unlike the United States Titan program, did not originate as a ballistic missile. It was designed from the beginning as a pure space launch vehicle. There had been much speculation that the Proton launcher was the CIS version of the Saturn type launcher for

a manned mission to the moon. Information released in 1989 concerning N1, the lunar mission class launcher, however showed that the Proton and its stages were not developed for this purpose. [Ref. 7:p. 253]

Design work on the Proton family of launch vehicles began in the early 1960's in the design bureau headed by V.H. Chelomey. The first variant of the family was the two stage Proton, or SL-9, or D version. The first stage was powered by six separate engines, while the second stage consisted of four engines. The fuel used for the original D variant consisted of the high-temperature propellant components nitrogen tetroxide (N_2O_4) and asymmetric dimethylhydrazine (UDMH). [Ref. 10:p. 1] The engines of the first stage each had a separate fuel tank with an additional larger tank located so as to crossfeed each engine. This was the same basic design which was used on the United States Saturn 1B. [Ref. 5:p: 132] The first Proton vehicle was launched on 16 July 1965 and carried the Proton 1 satellite into orbit. Only four SL-9 boosters were launched.

The remaining two versions of the Proton, SL-12 (D-1-e) and the SL-13 (D-1) are still in use today by the CIS. Both of these version still use the core Proton stages. Figure 8 shows the SL-12 and SL-13.

The D-1-e, which is the four stage version, was first successfully launched on 10 March 1967. It was developed for

an eventual circumlunar flyby by a manned spacecraft. [Ref. 11:p. 34] The manned flyby of the moon was never completed. The launch vehicle itself consisted of the same two stages as the original Proton with the addition of third and fourth stages. The fourth stage is known as Block-D, and is used for orbit transfer or payload escape. The fourth stage is powered by oxygen and kerosene. A wide variety of missions have been flown on the SL-12, including geosynchronous communications satellites, such as Ekran, Raduga and Gorizont, and interplanetary missions, which included Mars and Phobos to Mars, Venera to Venus, and Zond and Luna to the moon. The D-1-e has had the most launches of all the Proton variants. [Ref. 5:p. 133]

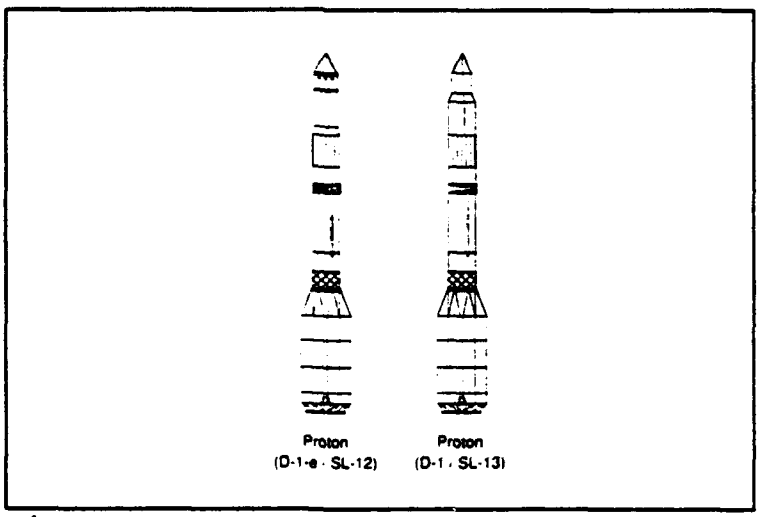


Figure 8. The Proton Launchers (SL-12 and SL-13) [Ref. 5:p. 132]

The third variant is the SL-13, or D-1. It is made up of the first three stages which comprise the SL-12. It was

first flown successfully in 1968 and has been used extensively to place heavy payloads into LEO. These payloads include all of the Russian space stations from Salyut-1 in 1971 through Mir in 1986. [Ref. 5:p. 132]

Proton has been considered operational since 1970. The boosters are assembled at the Krunitschev factory near Moscow by the KB Salyut design bureau. Production of the Proton was reduced from 12 vehicles a year to eight a year in 1989. [Ref. 7:p. 253] The Baikonur Cosmodrome, outside of Tyuratam, is the location of all the Proton launches.

2. Success Rate

The Proton family of launchers suffered a tumultuous beginning, experiencing a number of failures in its early years of existence, but as the system matured it became more and more reliable. The overall success rate of the Proton family of launchers is 85.4% through the end of May 1993. Figure 9 shows the Protons launch success rate.

SL-9 had a 75% success rate with one vehicle out of the four not reaching orbit. The second stage of the third launch of the Proton failed and resulted in the loss of the Proton-3A satellite.

Of the 210 Protons launched from July 1965 through May 1993, 179 of them were the D-1-e version. The success rate for the SL-12 is 86.6% with 24 failures out of the 179 attempts. The year 1969 was particularly bad for the SL-12;

of the eight attempted launches during that year six of them failed. Stage 2 was responsible for 50% of those failures. In the last ten years, the D-1-e has had only four failures in 100 attempts for a 96% success rate. [Ref. 8:p. 182]

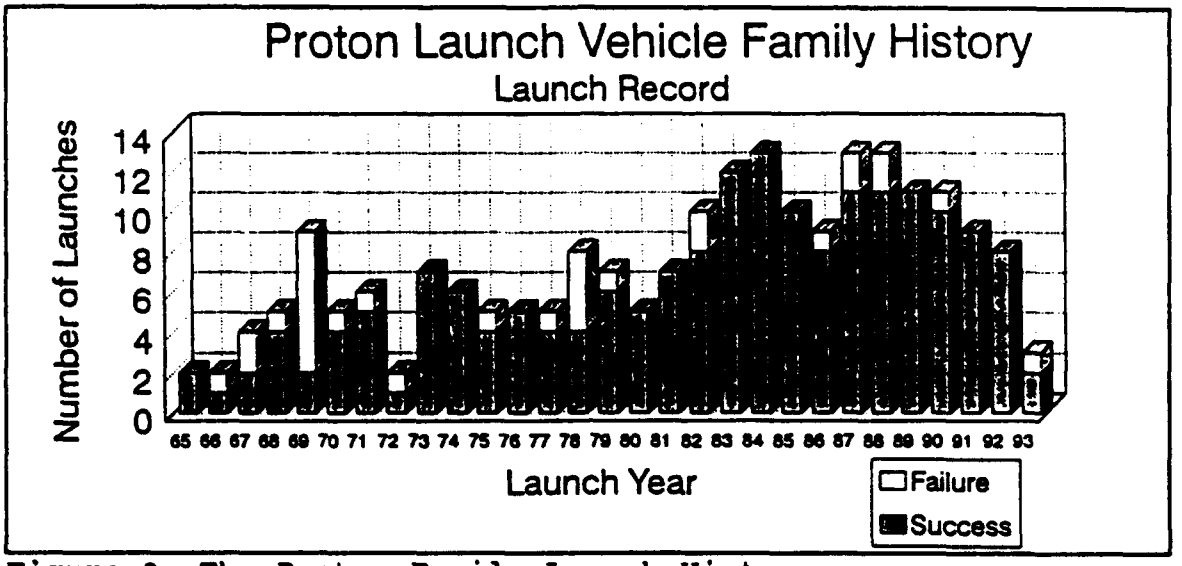


Figure 9 The Proton Family Launch History.

The D-1 version of the Proton has had a 81.5% success rate since its first flight in 1968. Of the 27 missions attempted using the SL-13 five of them have failed. Three of those failures have been attributed to the second stage.

Here again it should be noted that all the flights from 1965 through 1969 were considered to be test flights by the CIS. This means that, of the 30 failures in the Proton history, 11 of them occurred during the testing stage.

B. PROTON VEHICLE DESCRIPTION

There are currently two of the original three versions of the Proton still in production. The current versions are the SL-12 and the SL-13. For the purpose of the vehicle description the SL-12 will be utilized since it is basically the SL-13 with the addition of a fourth stage.

The complete vehicle is 197 feet tall with a gross mass of 1,550,000 lb. The D-1 can place 44,100 lb into a 100 nm orbit (LEO), while the D-1-e can place 4,850 lb into a geosynchronous orbit.

1. Stage 1

The first stage of the Proton launch vehicle is 66.3 feet tall and has a maximum diameter of 24 feet. This stage weighs 1,004,000 lb of which 904,000 lb is fuel. An aluminum alloy is used in its construction. Figure 10 shows the Proton's first stage.

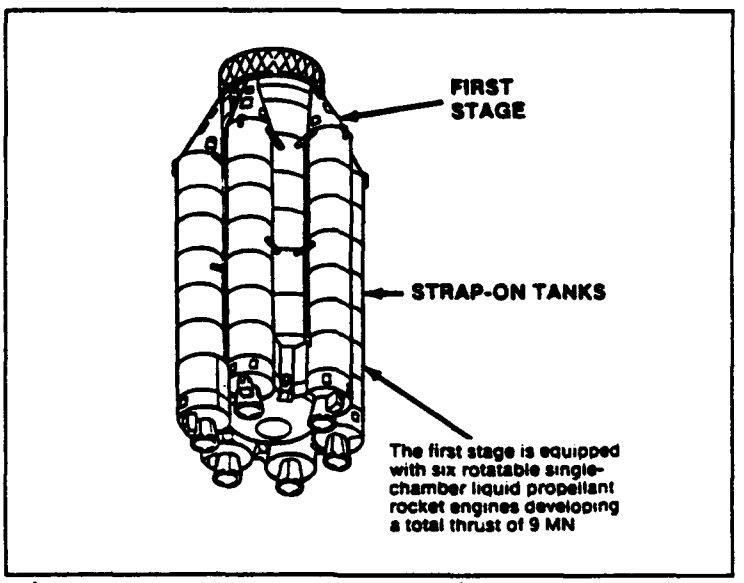


Figure 10. The Proton First Stage.
[Ref. 5:p. 136]

There are six RD-253 engines in the first stage. The engines are symmetrically placed about a center oxidizer tank. Each engine is capable of producing 392,291 lb vac of thrust in the new uprated version, with an Isp of 317 seconds vac. [Ref. 8:p. 242]

The propellant tanks, one for each engine, are internally coated with an anti-corrosive coating. The main oxidizer tank is protected from corrosion by the mixing of a abator with the oxidizer. [Ref. 5:p. 137] The center oxidizer tank feeds all the main engines through crossfeed piping.

The first stage burns for approximately 130 seconds, and is a burn-to-depletion booster. The first stage has no restart capability. Roll, pitch and yaw are controlled through the gimbaling of the six main engine nozzles. This gimbaling provides for the rotation of the separate engine compartments in a plane which is parallel to the longitudinal axis of the booster. [Ref. 5:p. 137]

2. Stage 2

The second stage of the SL-12 is 45.0 feet tall and 13 feet in diameter. This stage weighs 365,000 lb of which 330,000 lb is fuel. Like the first stage it is also made from aluminum alloy. Figure 11 shows the second stage.

Propulsive power for the second stage comes from four RD-0210 engines; three are the RD-465 version while the fourth is single RD-468 version. These engines use the same fuel as

the six first stage engines. Fuel storage is much different from the first stage. In the second stage there are two fuel tanks placed in tandem, and they share a common bulkhead. Piping from the top tank runs through the center of the lower tank to feed each of the engines. These engines produce 131,063 lb vac of thrust each with an Isp of 327.4 seconds vac. [Ref. 8:p. 267]

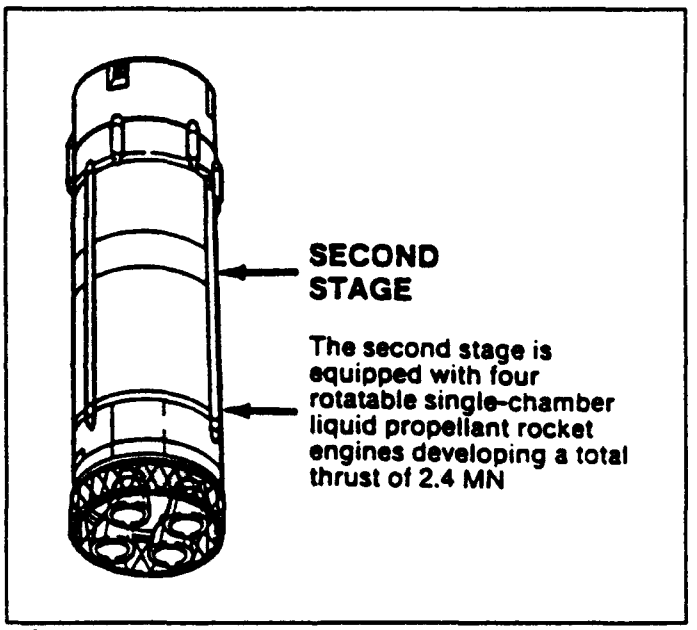


Figure 11. The Proton Second Stage. [Ref. 5:p. 136]

The second stage burns for approximately 212 seconds and is also a burn-to-depletion stage. There is no restart capability on this stage. There is no attitude control built into this stage. Attitude control is controlled by the verniers on the third stage.

3. Stage 3

The third stage of the D-1-e is 30 feet tall and has a diameter of 13 feet. It has a gross weight of 123,000 lb of which 110,000 lb is fuel. Like the previous two stages it is made from an aluminum alloy. This stage will place the fourth stage and payload into a 200 km circular parking orbit about ten minutes after launch. Figure 12 shows the third stage.

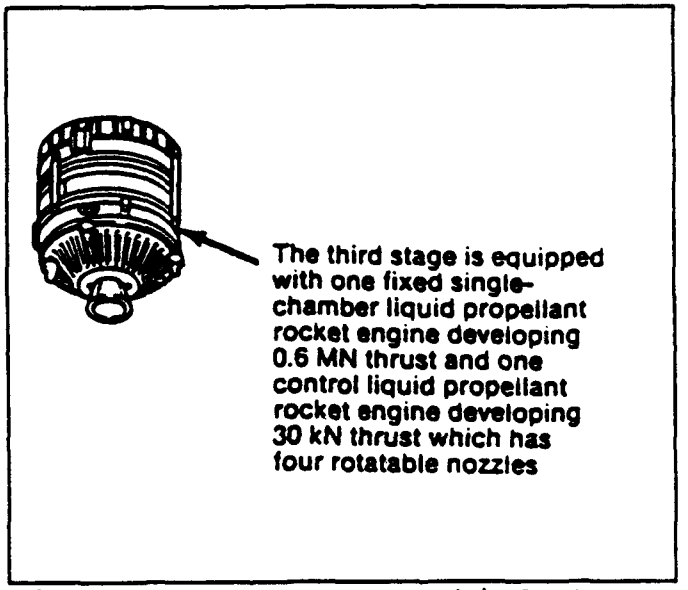


Figure 12. The Proton Third Stage. [Ref. 5:p. 136]

A single RD-0210 type engine, similar to the ones used in the second stage, provides 134,660 lb vac of thrust at an Isp of 325.3 seconds vac. [Ref. 8:p. 267] The engine designation for this stage is the RD-473 which is a variant of the RD-0210 engine family. This engine burns the same type of fuel used by the previous ten engines of the Proton.

The third stage can burn for between 250 and 350 seconds. It has no restart capability, but the burn time can be varied to achieve the required velocity for orbit. Attitude control is provided by four gimbaled verniers which provide an additional 6,969 lb vac of thrust. [Ref. 4:p. 137] This is the basic composition of the SL-13. The next stage is what separates the SL-12 from the SL-13.

4. Stage 4

The fourth and final stage of the D-1-e is 18 feet tall and has a diameter of 11.5 - 13 feet. This stage was originally designed as the N1's fifth stage. It has a gross weight of between 38,900-44,000 lb of which the fuel weighs 33.000-38,000 lb. [Ref. 5:p. 137] It is constructed from an aluminum alloy. The fourth stage is shown in Figure 13.

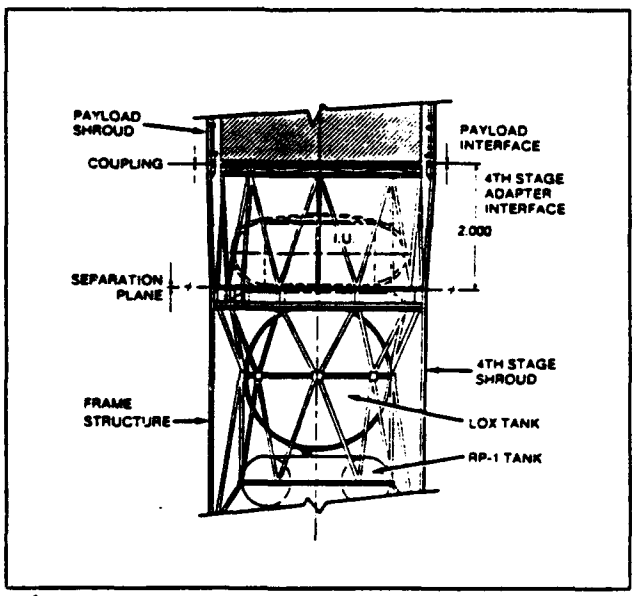


Figure 13. The Proton Fourth Stage. [Ref. 5:p. 136]

There are two versions of this stage, the Block D and Block DM. The Block DM version is used to launch Glonass and GEO missions and carries the Proton's control systems for their launches. The Block D version requires that the payload provide that control. [Ref. 8:p. 255]

The fourth stage is powered by a single 58M engine. There appear to be two different fuels used by this stage, the standard fuel being kerosene and the other fuel being sintin[1]. Sintin should produce a higher Isp. The kerosene version provides for 19,108 lb vac of thrust while the sintin version produces 18,771 lb vac of thrust. [Ref. 8:p. 256] The tank arrangement for this stage is again a tandem configuration with a toroidal shaped tank on the bottom and a spherical tank above it. The lower tank contains the fuel, kerosene or sintin, while the upper tank contains liquid oxygen (LOX). The LOX tank is thermally insulated. The engine is capable of being restarted seven times within a two day period; five engine restarts have been accomplished on a single mission. The maximum burn time for the fourth stage is approximately 680 seconds.

C. PAYLOAD SECTION

There are currently three different payload fairings available for use with the D-1-e. They vary in overall height

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[1] Sintin is known only as a hydrocarbon-based fuel.

from 299 inches up to 447 inches. The payload sections are shown in Figure 14.

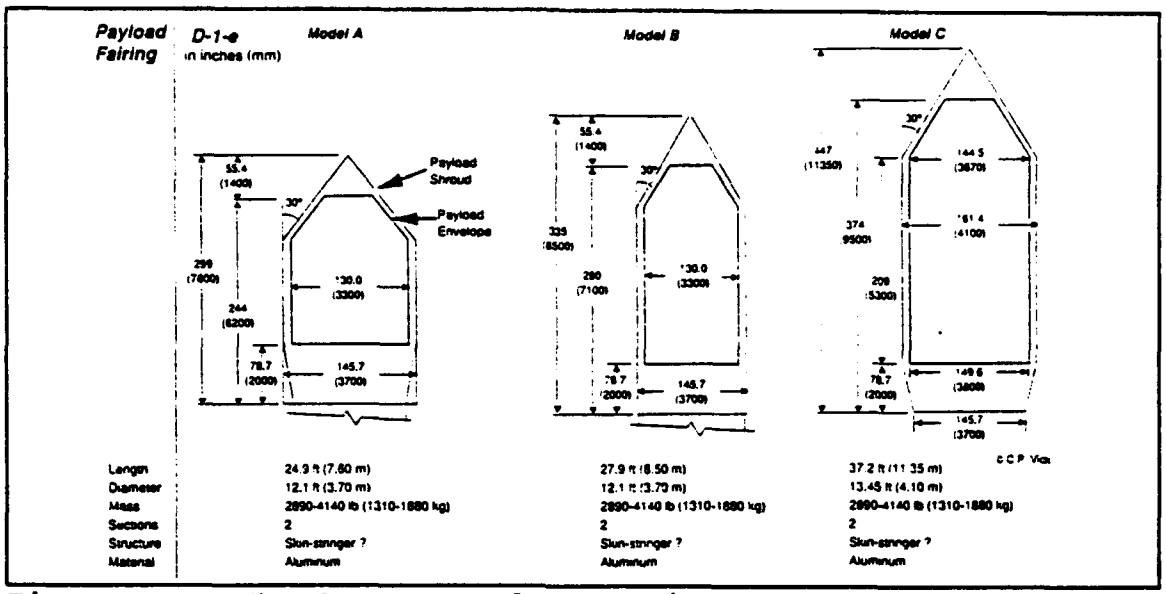

Figure 14. The Proton Payload Options. [Ref. 5:p. 138]

The smallest payload fairing is the Model A version. Its outside dimensions are 299 inches (24.9 feet) tall and 145.7 inches (12.1 feet) in diameter. The payload envelope is approximately 165.3 inches tall and 130.0 inches in diameter. The fairing is made up of two pieces. The fairing is insulated both inside and outside, with additional acoustic insulation available.

The medium size fairing has external measurements of 335 inches (27.9 feet) and an external diameter of 145.7 inches (12.1 feet). The payload compartment on this model is 130 inches in diameter and 201.3 inches in height. As with the Model A it is also of two piece design.

The third and largest fairing, Model C, is 447 inches (37.2 feet) tall and has a maximum external diameter of 161.4 inches (13.45 feet). The payload section for this fairing is a maximum of 149.6 inches in diameter at the base and 295.3 inches tall. [Ref. 5:p. 138] As with the previous two fairings additional insulation is available for the protection of the payload. This fairing is of a two piece design as well.

All the fairings are attached to the fourth stage cylinder shroud of the D-1-e model and are constructed from aluminum. The primary function of the fairing is to protect the payload from thermal and aerodynamic loads during launch and prior to orbital insertion. The fairing is jettisoned as the launch vehicle passes through 78.7 nm, which occurs approximately 370 seconds after launch with the vehicle traveling at 2.4 nm/s. [Ref. 12:p. 1]

D. TYPICAL MISSION PROFILE

All Proton missions are flown from the Baikonur Cosmodrome, located at Tyuratam, Kazakhstan. The Cosmodrome extends about 100 mi from east to west and about 55 mi from north to south. Tyuratam is the only facility in the CIS able to launch the Proton, Zenit, and Energia.

A typical D-1-e mission to place a payload into a geostationary orbit will take approximately seven hours and ten minutes, from launch to satellite separation from the

fourth stage. Launch preparation starts approximately 60 days prior to launch with the delivery of the payload and support equipment to the cosmodrome. At T-6 days, everything is ready for the actual mating of the payload to the launch vehicle.

At T-6, the payload is mated to the booster, the entire vehicle is placed on the erector transporter and is rolled out to the launch pad. Four hours are required to erect the complete vehicle on the pad. Continuous checking of the launcher and payload continue on through the final day of launch. [Ref. 5:p. 142]

At T-8:00 (Hours:Minutes), access is no longer permitted to the payload directly, although there is limited access to the payload up until T-1:30 through various access doors on the fairing.

Final fuel topping off for the first stage is completed at approximately T-0:45-0:30. At T-10 seconds, primary ignition occurs. First stage engines are brought up to medium thrust at T-4 seconds. Lift-off occurs at T-0 sec.

At time 0:00:00 (Hours:Minutes:Seconds), the main engines are brought up to nominal full thrust and the vehicle lifts off. A fraction of a second after liftoff, the liftoff service mechanism, which rises with the vehicle, is retracted. The liftoff service mechanism is a device which provides tracking information during the first fraction of powered flight and allows for the correction of the vehicle onto the

pre-planned flight profile. The roll program is initiated at 0:00:21.

Maximum dynamic pressure on the launch vehicle occurs at 0:01:00. Second stage engines are brought up to medium thrust range at 0:02:02; first stage shutoff and separation occur at time 0:02:07. At 0:03:03 the nose cone can be jettisoned, but usually is not. The steering engines on the third stage are brought on line at 0:05:34. Three seconds later the second stage engines are shut down, followed immediately by second stage separation. At 0:05:41, the third stage engine is fired. [Ref. 5:p. 143]

The nose cone is jettisoned and the shroud fairing separates at 0:06:10. After completing a burn of approximately 3 minutes and 57 seconds the third stage is commanded to shutdown. At 0:09:57 the third stage is separated. For the remainder of the flight the fourth stage is making numerous orbit-correcting burns to place the satellite in its proper orbit prior to payload separation which normally occurs at 7:09:36, marking the end of the Proton mission. [Ref. 5:p. 143]

E. COST

On 15 September 1993, Lockheed-Khrunichev-Energia International (LKEI) announced that it has signed its first launch service agreement for the Proton launcher. The agreement was signed with Space Systems/Loral, Palo Alto,

California and was for the launching of up to five Loral-built satellites on the Proton launch vehicle. The first launch is scheduled for the last quarter of 1995. Lockheed Chairman Dan Tellep stated, "With our combined capabilities, LKEI can offer satellite manufacturers and other customers reliable, cost - competitive access to space." [Ref. 13:p. 1]

The estimated price for launching the D-1-e, with a geosynchronous payload, is 35-70 million 1991 U.S. dollars. [Ref. 5:p. 135]

IV. ENERGIA

A. HISTORY

1. Development

Development of the SL-17, Sheldon designation K-1, vehicle began in 1974. It is widely believed that the failure of the Soviet Moon program based on the N-1, which contained 30 engines in the first stage, was the driving force behind the development of the Energia[2]. The N-1 was designed to produce 10.1 million pounds of total thrust.

The Energia was designed to be a modular, heavy lift launch vehicle. It is modular since, the number of strap-on boosters could be varied, and a wide variety of payloads could be used. The Buran, the Russian space shuttle, is one of the primary payloads to be carried. [Ref. 8:p. 258]

The number of strap-on boosters can vary between four and eight. There was some discussion and investigation into developing a version with two strap-on boosters and a smaller core, designated the Energia-M, but that plan appears to be on hold due to the expense of the program. Figure 15 shows the two variants of the Energia.

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[2] Energia translates to "Energy". This is a change from the normal method of naming CIS launch vehicles.

The core stage of the Energia houses four engines and associated equipment. The Energia has been designed with a tremendous amount of redundancy and safety factors built in. The current production version of the K-1 can still attain orbit even if one of the strap-on boosters shuts down or even if one of the core engines shuts down.

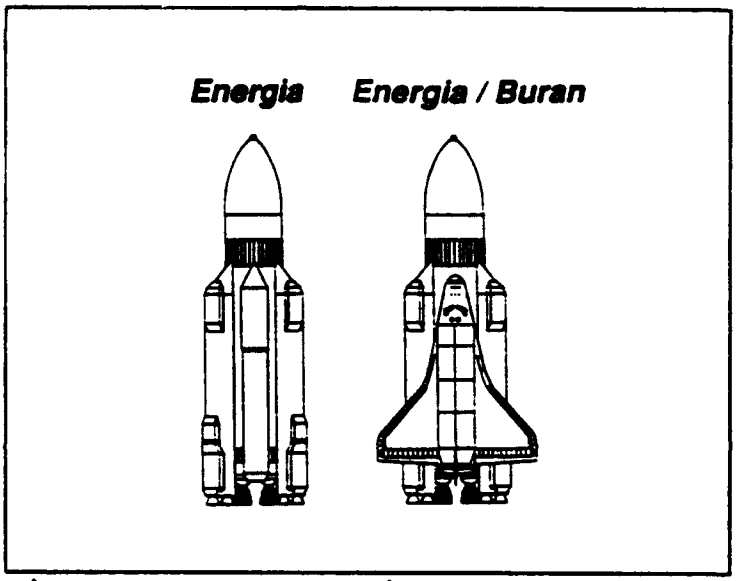

Figure 15. The Energia. [Ref. 5:p. 108]

At liftoff the K-1 has a gross weight over 2000 tons. It has a payload capacity of 100 tons into LEO with four strap-on boosters. [Ref. 14:p. 1] With the addition of four more strap-on boosters the Energia is capable of putting up to 200 tons into a low earth orbit.

After 13 years of development at an estimated cost of 14 billion rubles, the first SL-17 successfully lifted off the launch pad from Baikonur Cosmodrome on 15 May 1987.

2. Success Rate

The success of the Energia launch vehicle is currently at 100%. There have been two successful launches of the Energia. The first was a launch of a mock up of a satellite and the second was a launch of the Buran. Figure 16 shows the Energia launch record.

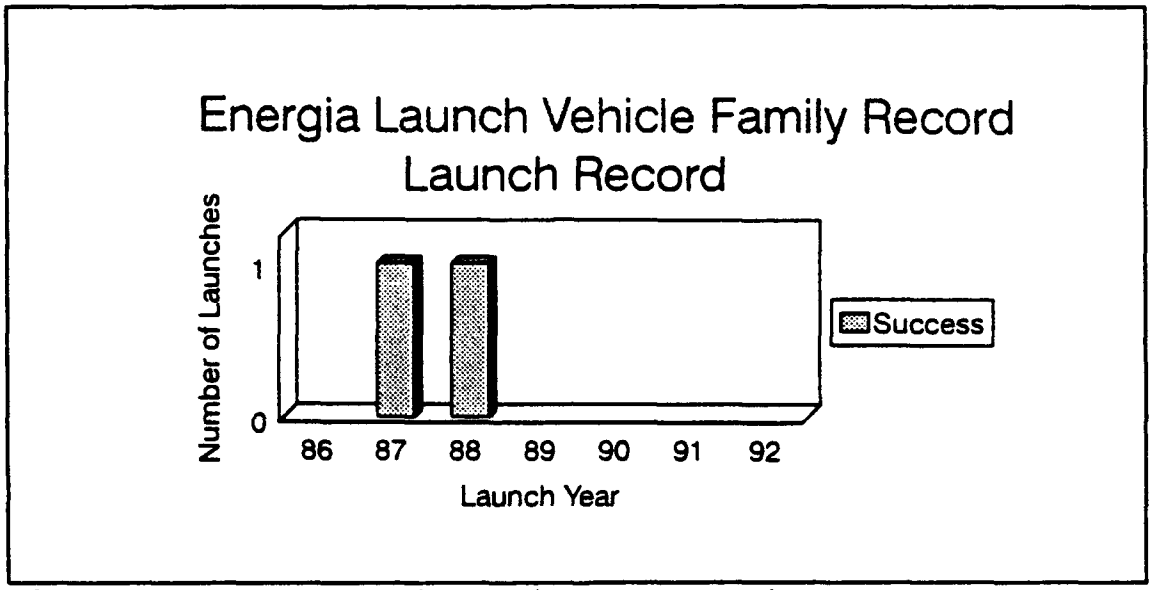

Figure 16. The Energia Family Launch History.

The first flight was termed a success. An announcement concerning the launch said that the first stage landed, as planned, in the Soviet Union and that the second stage delivered its payload as planned. The payload was delivered but did not go into orbit, due to a malfunction of the onboard system of the payload, and it splashed down in the Pacific. The launch announcement claimed "the aims and objectives of the first launch have been fully met." [Ref. 5:p. 107]

The second flight of the Energia, in 1988, was the successful launching of the Buran. The Buran has a launch mass of 105 tons, with a 30 ton payload capacity. The Energia launch vehicle burned its engines for 460 seconds and the Buran was separated from the Energia at 100 km after eight minutes of flight. [Ref. 8:p. 134]

The Energia is a modern heavy lift booster that the Russian government has invested a significant amount of time and money in. The safety features, which have been incorporated at all levels, potentially make the Energia one of the safest and most reliable launchers of all time.

B. ENERGIA VEHICLE DESCRIPTION

In a presentation given to the 39th Congress of the International Astronautical Federation, during 8-15 October 1988, in Bangalore, India, Dr. B.I. Gubanov, Glavkosmos, Moscow described the Energia follows:

> The launch vehicle has a two-stage configuration with parallel arrangement of rocket stages and side allocation of the payload. While developing this launch vehicle, the latest scientific and technical achievements of the Soviet rocket manufacturing were used. [Ref. 14:p. 2]

The Energia is designed with a tremendous amount of reliability. This is done through the incorporation of numerous redundant systems. The design criteria for the SL-17 was that one failure in any one system would not affect the fulfillment of the program, and that a second failure in the same system would not affect the safety of flight. [Ref. 10:p.

193] An example of the redundancy provided is in the turbogenerator power supplies, where there is not simple redundancy but quadruple redundancy. Other examples include the doubling of the batteries in the strap-on boosters and a doubling of the separation devices. [Ref. 5:p. 110]

1. Stage 1

The strap-on boosters, which are employed in pairs, are 131 feet high and have an outer diameter of 12.8 feet. Each booster weighs 783,000 lb of which 705,000 lb is fuel. The RD-170 engine provides the propulsive force for each booster. The boosters are made out of aluminum. Figure 17 shows the strap-on boosters.

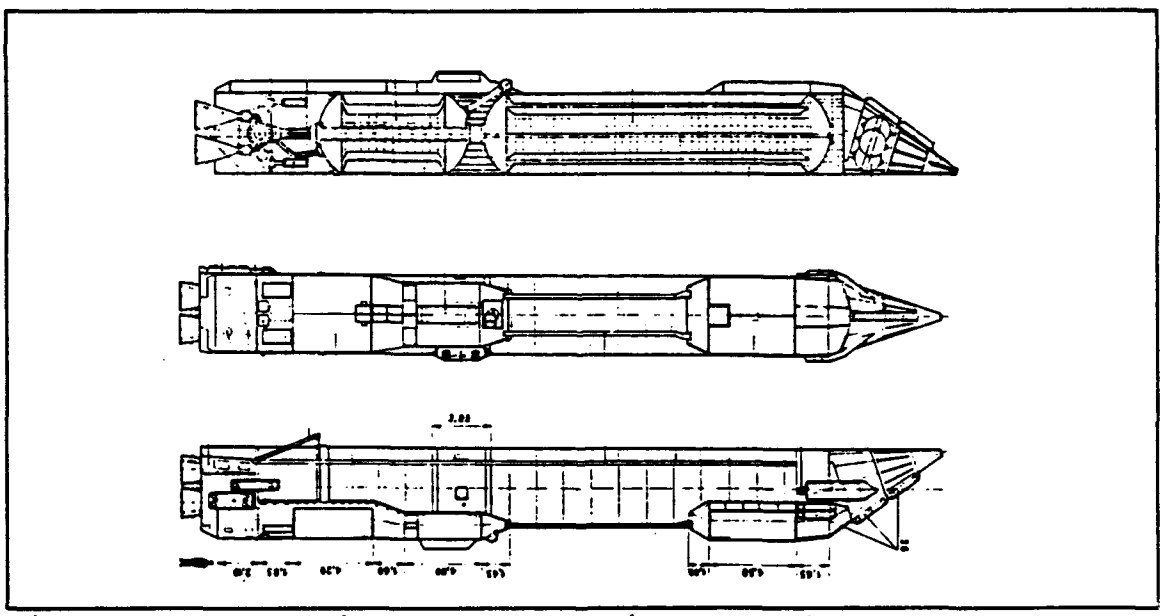

Figure 17. Energia Strap-On Engines. [Ref. 5:p. 109]

The RD-170 was developed by NPO Yuzhnoye, and a similar model was developed concurrently and is used to launch

the Zenit booster. The RD-170 is the highest thrust liquid propellant rocket engine ever flown. [Ref. 8:p. 241]

The engine is fueled by liquid oxygen and kerosene, which are stored in two separate tanks in the strap-on booster. Each engine is capable of developing 1,777,000 lb vac thrust with an Isp of 337 seconds vac. Each RD-170 engine consists of a single turbopump and four chambers. Through 1992 the engine design had been through more than 900 test firings and had more than 100,000 seconds of burn time. The engines themselves are produced in batches of five. One of those five is taken through three life cycles, while the remaining four engines undergo the standard acceptance tests. [Ref. 8:p. 241]

The first stage engines burn for approximately 145 sec and are burn-to-depletion engines. There is a throttling capability between 49 and 102% with this engine. The design of the engine allows smooth application of full power in 2 seconds. The strap-on engines have no restart capability.

Attitude control is provided through gimbaled nozzles which provide control in the pitch, roll and yaw axes, within $\pm 5°$. [Ref. 5:p. 110]

2. Stage 2

The second stage of the Energia is also referred to as the core stage. It is 197 feet high and has a diameter of 26 feet. The core has a gross weight of 1,995,000 lb; fuel

comprises 1,810,000 lb of that weight. The primary construction material for this stage is aluminum. Figure 18 shows the Energia core stage.

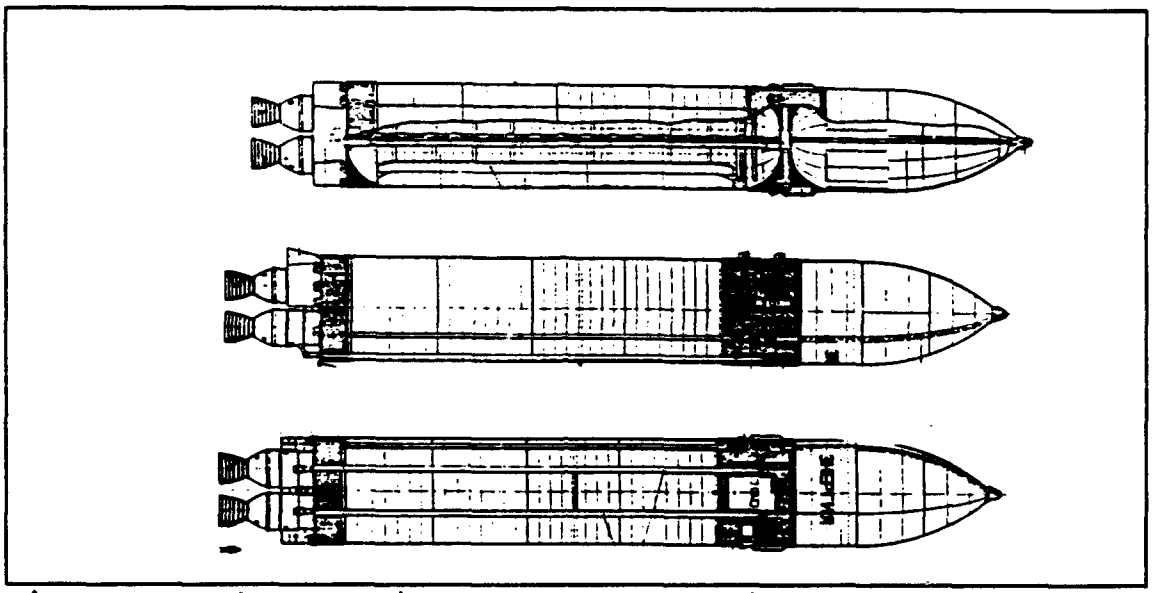

Figure 18. The Energia Core Stage. [Ref. 5:p. 109]

The core of the K-1 is powered by four RD-0120 engines. These engines provide 441,000 lb vac of thrust each with an Isp of 452.5 seconds vac. These engines are the first cryogenic engines used by the CIS. The fuel used is liquid hydrogen with liquid oxygen as the oxidizer. The engines have a nominal burn time of 480 seconds with a operational maximum of 600 seconds. Over 800 test firings of the RD-0120 have been accomplished with over 166,000 seconds logged. [Ref. 8:p. 245] There is no restart capability of the core engines, and they operate on the command shutdown principle.

Attitude control is provided through gimbaled nozzles. The gimbaled nozzles allow for ± 11° of attitude control.

C. PAYLOAD SECTION

The first launch of the Energia saw the failure of its payload section. While the first and second stages performed as expected, the kick stage failed to provide the required 100 meter per second velocity required to place the payload into orbit. [Ref. 7:p. 259] Figure 19 shows the cargo carrier for the Energia.

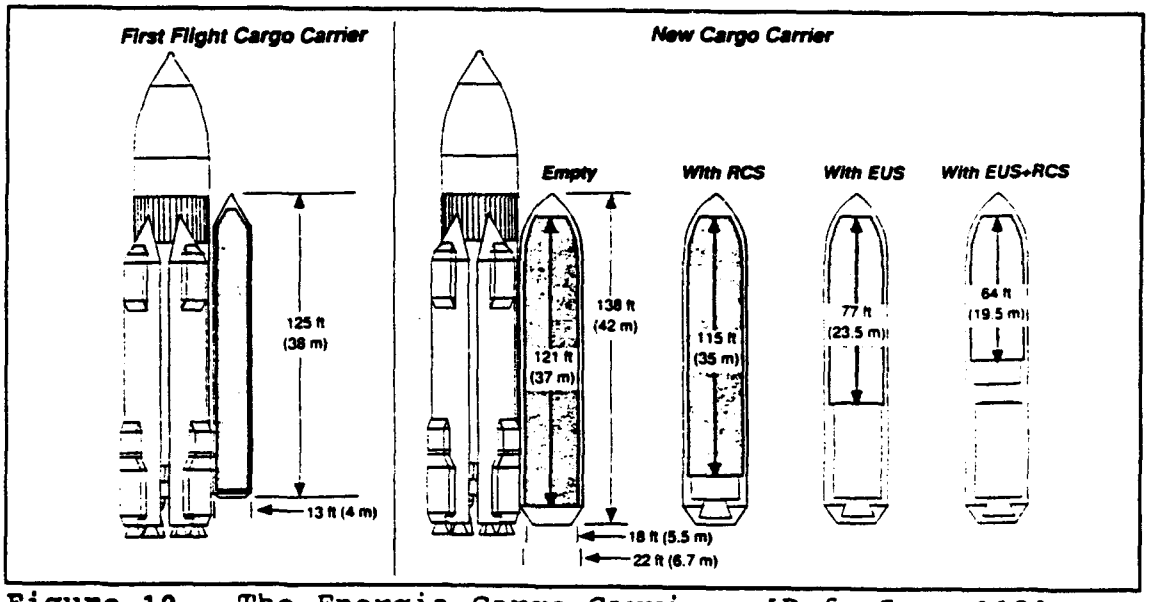

Figure 19. The Energia Cargo Carrier. [Ref. 5:p. 112]

The payload section of the K-1 is currently designed to be a side-mounted carrier. There has been discussion in developing a tandem mounted carrier which would allow for the launching of extremely heavy payloads. Currently there are four versions of the cargo section being developed. These

models include a basic empty carrier, the same carrier with a retro and correcting stage (RCS), the same carrier with a Energia upper stage (EUS) and a combination EUS and RCS carrier.

The first flight of the Energia used a smaller than standard carrier which was only 125 feet tall and had a diameter of 13 feet. The kick stage which failed is believed to be similar to the one to be used on the Buran for orbital injections of its payload. [Ref. 7:p. 259]

The basic empty carrier stands 138 feet tall and has an outside diameter of 22 feet. The inside, payload area, is 121 feet tall and has a diameter of 18 feet. The addition of the RCS would reduce the vertical payload dimension down to 115 feet, while the use of the EUS would reduce it down to 77 feet. A payload that used the EUS and RCS combination would reduce the dimension down to 64 feet. The payload carrier itself is constructed of aluminum. [Ref. 5:p. 112]

The RCS would be used to place payloads of up to 18 tons into geosynchrous orbit, or up 105 tons into a low earth orbit, with the use of four strap on boosters at launch. The RCS itself is 18 feet in length and 12 feet in diameter. It has a gross mass of 37,000 lb of which 33,000 lb is fuel. The RCS engine burns a mixture of liquid oxygen and kerosene. The oxidizer is carried in a spherical tank located above a torus shaped tank which carries the fuel. The engine is capable of producing 19,100 lb vac of thrust. It has a burn time of

approximately 600 seconds and can be restarted seven times. [Ref. 7:p. 259]

The EUS is designed to place large payloads into high orbits and to act as the first stage for moon and planetary missions. The EUS is 54 feet long and has a diameter of 18.7 feet. It has a gross mass of 170,000 lb with fuel accounting for 154,000 lb. Like the core stage of the Energia this is also a cryogenic stage. The engine burns liquid oxygen and liquid hydrogen. The main engine of this stage produces 16,860 - 22,480 lb vac thrust and is capable of ten engine starts. [Ref. 7:p. 259]

The final variant is the combined EUS and RCS. The primary use of this combination is for interplanetary missions. The EUS would provide the post-boost power and control while the RCS would provide the final power and trajectory control. The components would be the same as described above, loaded in a tandem configuration with the RCS placed above the EUS.

D. TYPICAL MISSION PROFILE

With only two launches of the Energia completed not much is known as far as payload access prior to flight. Furthermore not much information is available concerning the preflight sequence.

All Energia missions are launched from one of three available pads at the Baikonur Cosmodrome. Two of the three

pads are specifically designed to be used with the Energia/Buran combination. Figure 20 shows a typical Energia flight sequence.

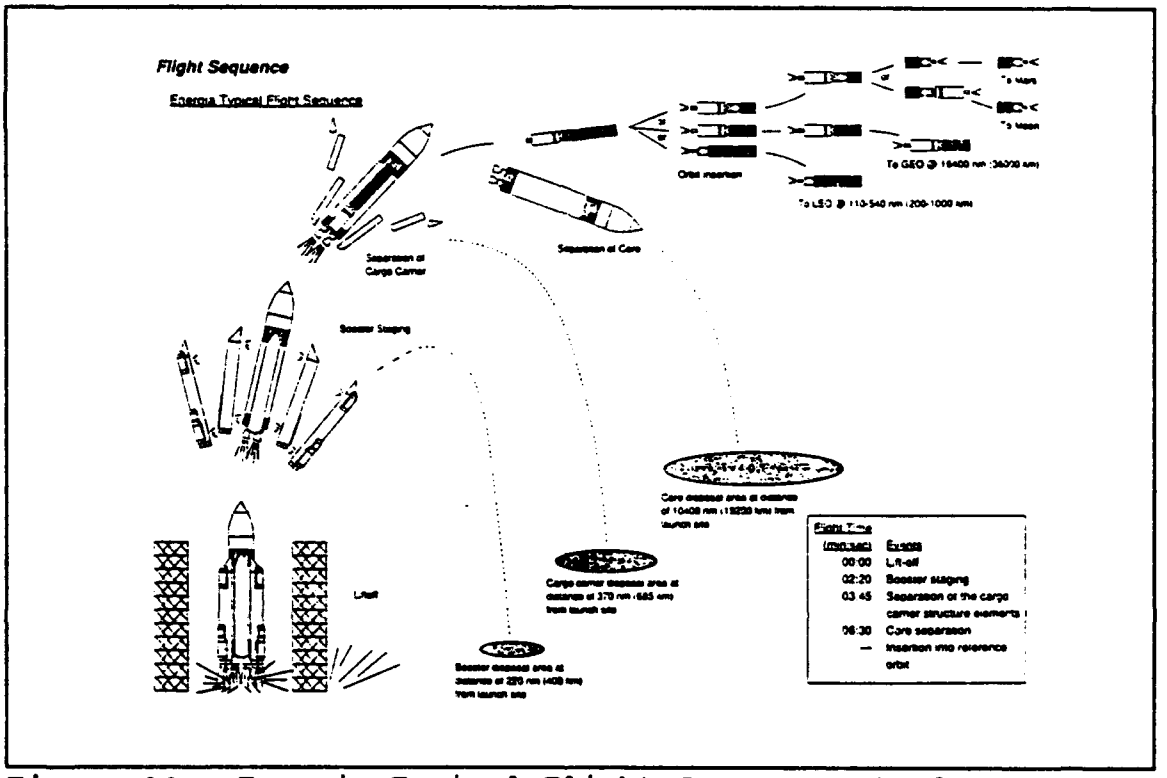

Figure 20. Energia Typical Flight Sequence. [Ref. 5:p. 116]

Fueling of the cryogenic stage occurs just prior to launch. The fuel is kept in special tank farms located approximately 1.2 miles from the pad. During the fueling procedure personnel are kept nine miles away from the area. [Ref. 5:p. 115]

At T-12 seconds, the core engines of the Energia are brought on line for check out and run up to full power. At 00:00 the strap-on boosters are ignited and the vehicle lifts off. At approximately 02:20 the boosters are jettisoned and

fall to earth 220 nm downrange. At time 03:45 the cargo carrier support structure elements are jettisoned. The pieces fall to earth approximately 370 nm downrange. The core separates from the payload approximately 06:30 after launch and it falls to earth 10,400 nm from the launch site. From this point on the mission depends on the payload and type of insertion mechanism used. [Ref. 5:p. 116]

E. COST

There is much interest on the part of the former Soviet Union to get buyers for their space launch vehicles. The programs are expensive to operate, and as the fiscal belt tightening continues in Russia, the space industry is going to need to become more and more self sufficient.

B.I. Gubanov of Glavkosmos, Moscow, stated during the 2nd European Aerospace Conference on Progress in Space Transportation, held in the Federal Republic of Germany 22-24 May 1989:

> There are many tasks. Work on the use of the ENERGIA versatile rocket-space transport system capabilities has only begun, and it shows that a large number of new tasks on commercial, scientific, and other trends of space exploration can already be solved in the near future. We are ready to cooperate on a mutually beneficial basis with all the countries interested in the solution of these problems[3]. Moreover, the solution of many of these problems is possible only by consolidation of efforts of a number of countries. [Ref. 10:p. 2]

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[3] The problems referred to include global communication systems and world ecological monitoring systems.

The cost of launching the K-1 is estimated to be 110 million U. S. dollars. This is based on 1991 figures. [Ref. 5:p. 108]

V. ARIANE

A. HISTORY

The Ariane family of launch vehicles are the result of a combination of three separate organizations, the European Space Agency (ESA), Centre national d'Etudes Spatiales (CNES), and Arianespace.

The ESA was established in December 1973 and is composed of 13 member nations, one associate member and one cooperating state. The 13 members are: Austria, Belgium, Denmark, France, Germany, Ireland, Italy, Netherlands, Norway, Spain, Sweden, Switzerland, and the United Kingdom. Finland is the associate member and Canada is the cooperating state. The primary purpose of ESA is the promotion of the peaceful use and exploration of space through cooperation between member nations.

CNES was created in 1962, by the French government, to promote and develop a French space program. It contributes technology but, more importantly, the launch facilities and launch coordination for the Ariane program.

Arianespace is a commercial venture which was established in 1980. The primary shareholders in Arianespace are France-58.5%, Germany-19.6% and Belgium-4.4%. The primary purpose of

Arianespace is for the commercialization of the Ariane launch vehicle. [Ref. 15:p. 28]

The Ariane 5 launcher is an entirely new launch vehicle from its predecessors. It is a departure from the evolutionary step-by step growth of the Ariane 1 up to the Ariane 4. This is because this marks the first Ariane launch vehicle to use a cryogenic core first stage. [Ref. 16:p. 3]

The predecessors to the Ariane 5 will be studied to understand the development of the Ariane family of launchers and to better comprehend the origin of the Ariane 5.

1. Development

The Ariane 1 program was conceived in 1973. The total development and qualification of Ariane 1 covered a period of eight and one half years. It was designed to place a 4070 lb satellite into a geosynchronous transfer orbit (GTO). [Ref. 5:p.31] The first successful launch of the Ariane 1 took place on 24 December 1979.

Ariane 1 was a three stage booster with a liftoff thrust of 553,480 lb vac thrust. The first stage consisted of five Viking engines and burned a mixture of asymmetric dimethylhydrazine (UDMH) and nitrogen tetroxide (N_2O_4). The first stage burned for approximately 146 seconds. The second stage contained a single Viking 4 engine and burned the same fuel as the first stage. The second stage developed 163,211 lb vac of thrust for 136 seconds. The third stage was a

cryogenic stage, which burned liquid oxygen and liquid hydrogen. It consisted of a single HM-7 engine which was able to develop 13,713 lb vac thrust and had enough fuel to burn for 545 seconds. There were 11 Ariane 1 launches from 1979 through 1986. [Ref. 8:p. 140]

Development for the follow-on to the Ariane 1 was started in July of 1980. The need for the follow-on was based on the fact that for Arianespace to remain competitive, it would need to be able to launch two payloads at a time. This led to the development of Ariane 2 and 3.

Ariane 2 was a modified version of the Ariane 1, and Ariane 3 was a modified version of Ariane 2. The major differences between Ariane 1 and Ariane 2 was the increased thrust of the Ariane 2 engines over the Ariane 1 engines, a 25% increase in the fuel capacity of the third stage and the incorporation of the Sylda[4] structure. [Ref. 5:p. 30] The first stage of the Ariane 2 was still powered by four Viking 5 engines but the fuel was changed to NTO and a mixture of UDMH and hydrazine hydrate. The new liftoff thrust was 604,511 lb vac, almost a 10% increase. [Ref. 8:p. 140] The Sylda allowed for the carrying of a dual satellite payload. The first Ariane 2 mission was flown on 21 November 1987. A total of six have been launched.

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[4] Systeme de lancement double Ariane

Ariane 3 was the Ariane 2 with the addition of two strap-on boosters. Each booster added an additional 149,722 lb vac of thrust and burned for 28 seconds. The boosters ignited 36 feet in the air vice on liftoff, and were spring jettisoned. [Ref. 8:p. 140] There have been 11 flights of the Ariane 3 with the first one being on 4 August 1984.

The current production version of the Ariane family, Ariane 4, started development only two years after the development of the Ariane 2/3 launcher. The Ariane 4 is still based on the original Ariane 1 three stage design but has had numerous upgrades and improvements. Six variants of the Ariane 4 have been used. A variant is based on the number and type of boosters used. The six variants are as follows:

- Ariane 40 no strap-on boosters
- Ariane 42P 2 solid strap-on boosters
- Ariane 44P 4 solid strap-on boosters
- Ariane 42L 2 liquid strap-on boosters
- Ariane 44LP 2 solid/2 liquid strap-on boosters
- Ariane 44L 4 liquid strap-on boosters

The Ariane 4 family of boosters provide Arianespace with the ability to place up to 9,830 lb into a GTO. The variety of versions allows for the customization of launch vehicles to meet customers' needs at the lowest possible price.

There have been 28 launches of the Ariane 4, through 12 May 1993, since its first launch in 1988. All the launches for the Ariane launch vehicles have originated from the Guiana Space Center (CSG) in Kourou, French Guiana.

2. Success Rate

Through January of 1994, there has been a total of 63 launches of Ariane rockets. Of those 63 only six have failed. That gives Arianespace a 90.48% vehicle success rate. The success rate of the Ariane 1 launch vehicle is 81.8% with two failures in 11 launches. The two failures were not related, one being a first stage combustion instability problem and the second being a third stage turbopump failure. Figure 21 shows the Ariane family launch history.

Ariane 2 suffered a failure on its maiden voyage. The third stage failed to ignite and the payload was not successfully placed into orbit. This gave the Ariane 2 family a 83.3% success rate with only one failure in six attempts.

The Ariane 3 series of launchers had only one failure as well. This occurred during its fifth launch when the third stage failed to ignite. With ten successful launches this gave the Ariane 3 launcher a success rate of 90.9%.

The current version of the Ariane launcher, Ariane 4, has placed 33 out of 35 payloads into orbit, through January 1994. This gives the Ariane 4 family a 94.28% success rate. The most recent failure for Ariane 4 occurred on 24 January

1994 when Ariane's Flight 63 failed, due to the third stage overheating and shutting down. While this is only the second failure of the Ariane 4 series of boosters, it marks the largest single loss ever suffered by the space insurance industry. "This is a failure, but failure is a price you pay in the high technology adventure of space," said Aerospatiale Chairman Louis Gallois. [Ref. 17:p. 1]

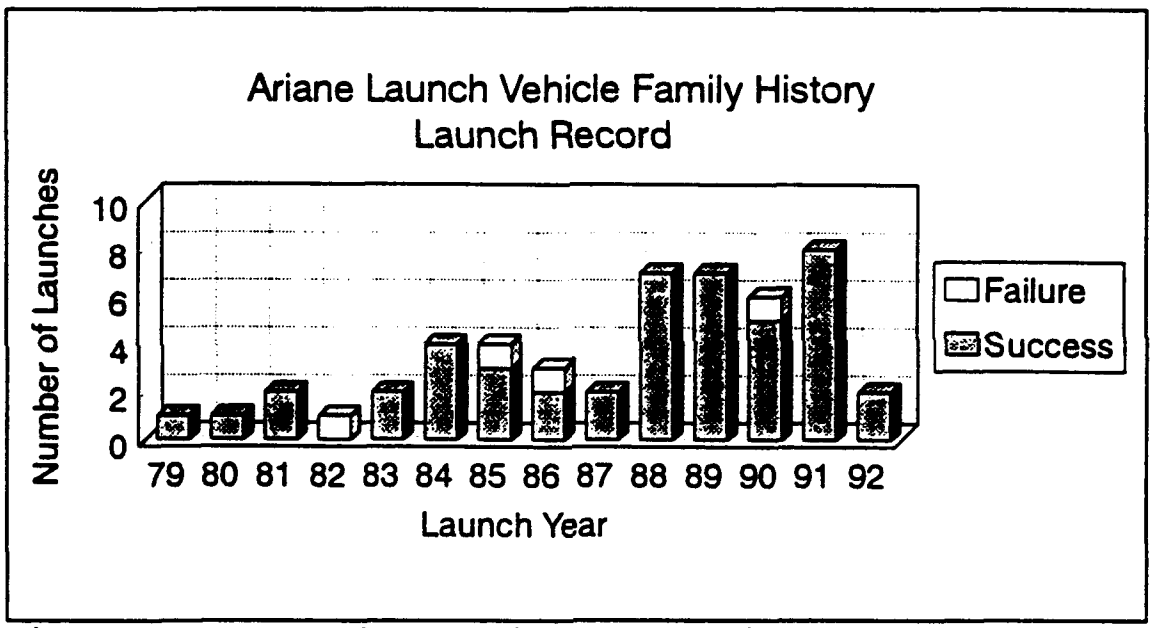

Figure 21. The Ariane Family Launch History.

B. ARIANE 5 VEHICLE DESCRIPTION

The latest addition to the Arianespace stable of Ariane launch vehicles will soon be the Ariane 5. The Ariane 5 marks the first time that Ariane will be using a cryogenic first stage.

Consideration for a follow-on to the Ariane 4 series of boosters started in 1984. 1988 was the year in which the design and development for the Ariane 5 began. There were six requirements that were decided upon; these were:

- Deliver into GTO one or more satellites with a total mass of 15,000 lb for a single launch configuration, or 13,000 lb in the dual launch configuration.
- Launching of the Hermes spaceplane, which weighs 48,500 lb, into a transfer orbit of 50 x 250 nmi.
- Be able to hold a payload with a 15 foot diameter.
- Have a reliability of 98%.
- Meet the safety requirement of 99.9% with the Hermes as a payload.
- Attain a 10% cost reduction of a dual payload launch when compared to the Ariane 44L. [Ref. 18:p. 203]

1. Booster Stage

The solid propellant, strap-on boosters to be used on the Ariane 5 are 98 feet tall and have an outer diameter of 9.94 feet. Each booster has a gross weight of 583,000 lb of which fuel makes up 506,000 lb. [Ref. 5:p 37] Each booster has a forward assembly consisting of an inclined cone which, through an attachment, transmits the trust to the core component and an aft skirt which is used to link the launch vehicle to the launch table. [Ref. 18:p. 204] Measurement systems, ignition systems, separation systems, recovery systems and destruction systems are all incorporated into the boosters.

Etage d'Appoint Poudre produces the P230 engine which are to be used as the motors for the Ariane 5 strap-on boosters. The P230 is capable of developing 1,180,246 lb vac of thrust using hydroxy terminated polybutadiene (HTPB) as a fuel. Figure 22 shows the Ariane strap-on stage.

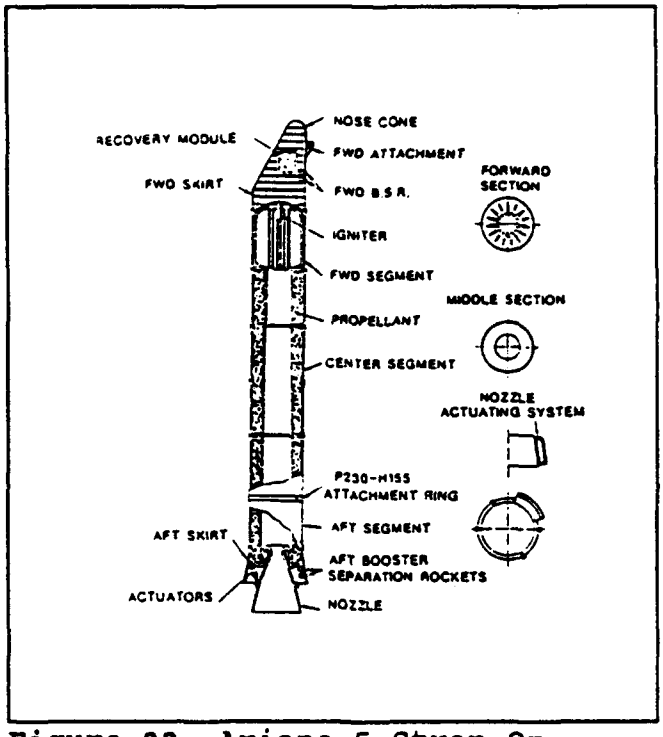

Figure 22. Ariane 5 Strap-On Stage. [Ref. 5:p. 37]

The first stage boosters are designed using monocoque construction and steel as the fabrication material. Each engine is designed to burn for approximately 123 seconds, with no restart capability. The boosters are designed to burn to depletion. Attitude control of $\pm$ 6° is provided through hydraulic gimbaling. [Ref. 5:p. 37]

2. Stage 1

The first stage of the Ariane 5 is the core stage, also known as H155. This is where the Ariane 5 differs the most from previous versions of the Ariane launcher family. The core stage is a cryogenic stage. Figure 23 shows the H155 stage.

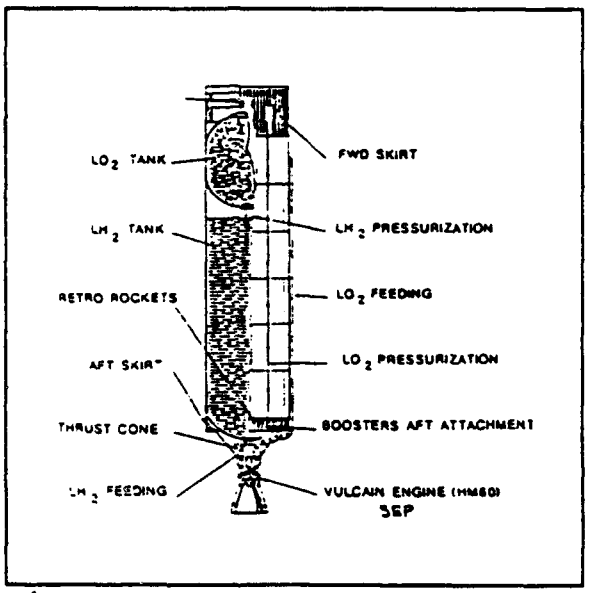

Figure 23. H155 Stage.
[Ref. 5:p. 37]

The H155 is 100.7 feet tall and has a diameter of 17.7 feet. It is made using semi-monocoque construction using aluminum 2219. The core stage has a gross mass of 375,000 lb of which the fuel and the oxidizer account for 342,000 lb. The main stage is composed of the H_2 tank, the O_2 tank, the thrust cone and both upper and lower skirts. [Ref. 19:p. 14]

The fuel tanks are cylindrical in shape with the O_2 tank located above the H_2 tank. The upper tank and the lower

tank share a common dome bulkhead designed to keep the two fuel components separated. The thermal insulation provided for the tanks comes in two parts, cold insulation and hot insulation. The cold insulation is used to keep the propellant in the required thermal state for use by the engine. The hot insulation protects the tanks from aerothermal fluxes, specifically the interaction fluxes from the solid boosters. [Ref. 19:p. 14]

The aft skirt provides the interface between the thrust cone and the tanks. The thrust cone transmits the thrust developed by the engine and supports the equipment for the propulsion plant. [Ref. 19:p. 14]

The first stage is powered by a single Vulcain engine. This engine is being designed to provide 257,406 lb vac thrust with an Isp of 430 seconds vac. There is no throttling capability and this is designed to be a command shutdown stage. [Ref. 5:p. 37]

The core stage will have a nominal burn time of 590 seconds, with no restart capability. Attitude control for the first stage is through hydraulic gimbaling which will provide $\pm 6^{\circ}$ of pitch and yaw control. Roll control will be provided using gas generators. [Ref. 8:p. 142]

3. Stage 2

The second stage of the Ariane 5 is to be used only on unmanned missions. When the Hermes is to be carried, the

second stage will be omitted and the manned section will mated directly to the first stage.

The second stage, designated as L7 (See Figure 24) or the storable propulsion stage (EPS), is only 14.8 feet high with a 17.7 foot diameter. It consists of four propellant tanks, helium bottles, actuators and a regeneratively cooled gimbaled engine. [Ref. 5:p. 137]

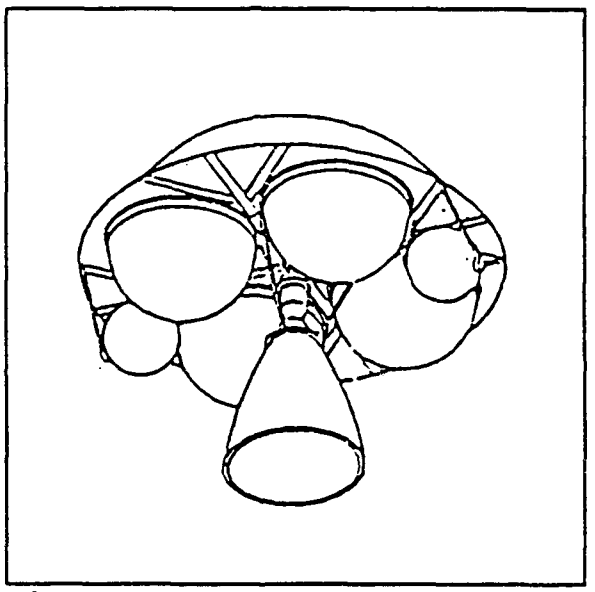

Figure 24. L7 Stage.
[Ref. 5:p. 37]

The engine uses N_2O_4 and MMH (monomethyl hydrazine) to produce 6,140 lb vac of thrust with an Isp of 316 seconds vac. The engine has a multiple restart capability and should also have some throttling capability. [Ref. 8:p. 144]

The second stage should be capable of a total burn time of 800 seconds. Attitude control for the second stage is

provided by the gimbaled engine in the pitch and yaw axis and through the use of the vehicle equipment bay (VEB) hot gas thrusters. [Ref. 5:p. 137]

The second stage is but one component of what is normally referred to as the Upper Composite. This is composed of the second stage, a VEB, and an upper section that consists of a short or long fairing, plus a Speltra[5] for dual or triple launches.

C. PAYLOAD SECTION

Three standard payload configurations are being considered for use with the Ariane 5 launcher. These three configurations include a single payload, a dual payload and a triple payload configuration. Figure 25 shows the three payload options.

There are two fairing configurations currently proposed for the Ariane 5. They are a 37.9 foot fairing and a 59.2 foot fairing. The longer fairing weighs 6400 lb while the shorter fairing weighs 4200 lb. The fairing is composed of sandwiched panels with an aluminum honeycomb core between CFRP (carbon-fiber-reinforced plastics) skin. [Ref. 5:p. 38]

Both fairing designs use a two piece configuration. The fairings are split down the vertical plane. The separation system proposed for use will be very similar to the one

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[5]Structure porteuse externe lancements triples Ariane.

currently in use on the Ariane 4 family of launchers. Separation of the fairing on the Ariane 4 starts when the clamp band holding the fairing is released. A pyrotechnic cord then cuts the fairing into two vertical halves and pushes them apart. [Ref. 5:p. 38]

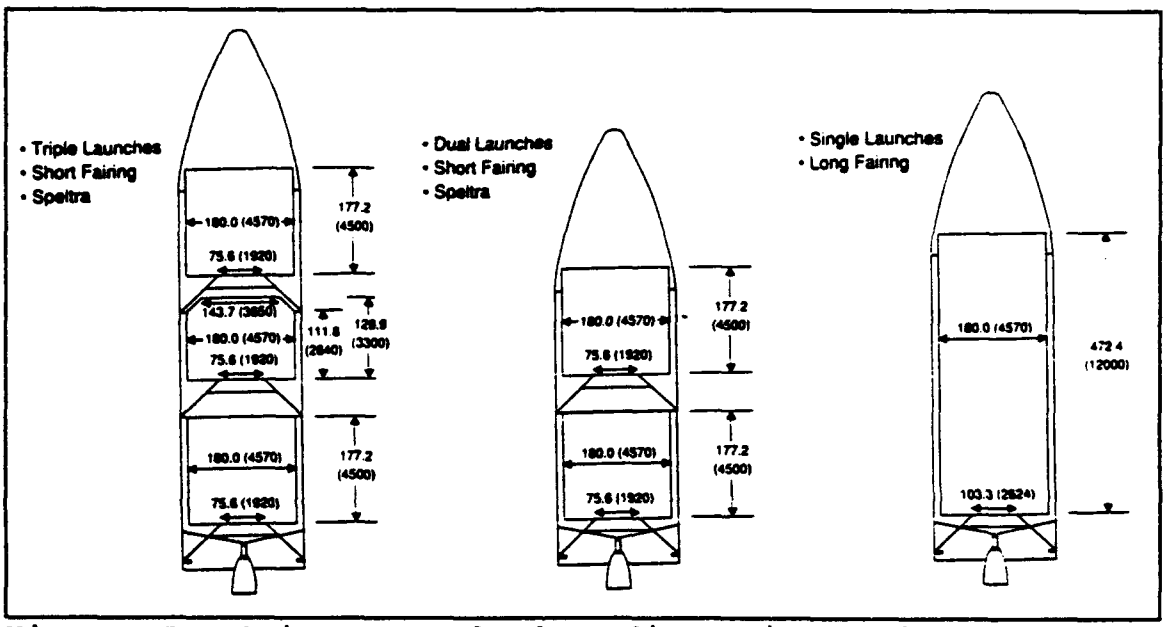

Figure 25. Ariane 5 Payload Configuration Options. [Ref. 5:p. 38]

When considering the payload section, the Speltra plays an important role. The Speltra is an external support structure which permits the launching of dual and triple payloads. It is made up of a cylinder interfacing with the VEB and the fairing or with another Speltra, and a cone carrying in its upper section a frame connecting it with the payload. The Speltra is composed of sandwiched panels, carbon fibre/resin composite skins covering an aluminum honeycomb core. [Ref. 19:p. 15]

D. TYPICAL MISSION PROFILE

There are six distinct phases which lead up to the liftoff of the Ariane 5 launcher. The phases all take place at the Ariane 5 launch site at the Guiana Space Center.

The first phase lasts 44 days and starts at the T-61 day point. This phase consists of preparation of the two solid boosters at the Booster Integration Building (BIP).

The second phase, which lasts approximately 10 days, is centered around preparation of the Core stage. This takes place in the Launcher Integration Building (BIL).

Phase three has two parts, the mating of the boosters with the core stage and a verification of the launcher without the payload. The first part occurs in the BIL and requires 3 days while the verification step takes 4 days and also occurs in the BIL.

The fourth phase of launcher preparation starts at approximately T-18 days. This is when the payload and nose fairings are prepared in the Final Assembly Building (BAF). The launcher is also transferred to the BAF during this stage. Nine days is the estimated time for the completion of this phase.

The fifth phase, which requires eight days, is composed of assembling the payloads and fairing on the launcher and preparing it for countdown.

The final phase starts at T-1.5 days before launch. The phase includes moving the launcher to the launch pad and final preparation and launch.

If a problem should arise during final countdown, with either the launcher or the payload, the entire vehicle must be rolled back into the launcher preparation zone. One of the new design features of the simplified Ariane launch sequence is to allow for the Ariane 5 to be rolled back defueled, moved back into position and refueled for a launch the following day. This is currently what is being requested by Arianespace. [Ref. 20:p. 46]

Figure 26 shows a typical Ariane 5 mission. Once the system is ready for launch and the countdown has reached 00:00 (minutes:seconds) the main engine of the core stage is ignited. At 00:03 the solid rocket boosters ignite and liftoff occurs. The maximum dynamic pressure occurs 9.32 miles into the flight at 01:11, maximum longitudinal acceleration occurs 32 seconds later at time 01:43 when the acceleration is 143.7 ft/sec^2. The strap-on boosters are jettisoned at 02:06 at an altitude of 34.8 miles. Fairing jettison follows at time 03:04 at an altitude of 65.9 miles. The cryogenic core stage burns out at an altitude of 87.6 miles and separation follows. EPS, second stage, flight ends at mission time 23:10 at an altitude of 669 miles with payload separation. [Ref. 5:p. 44]

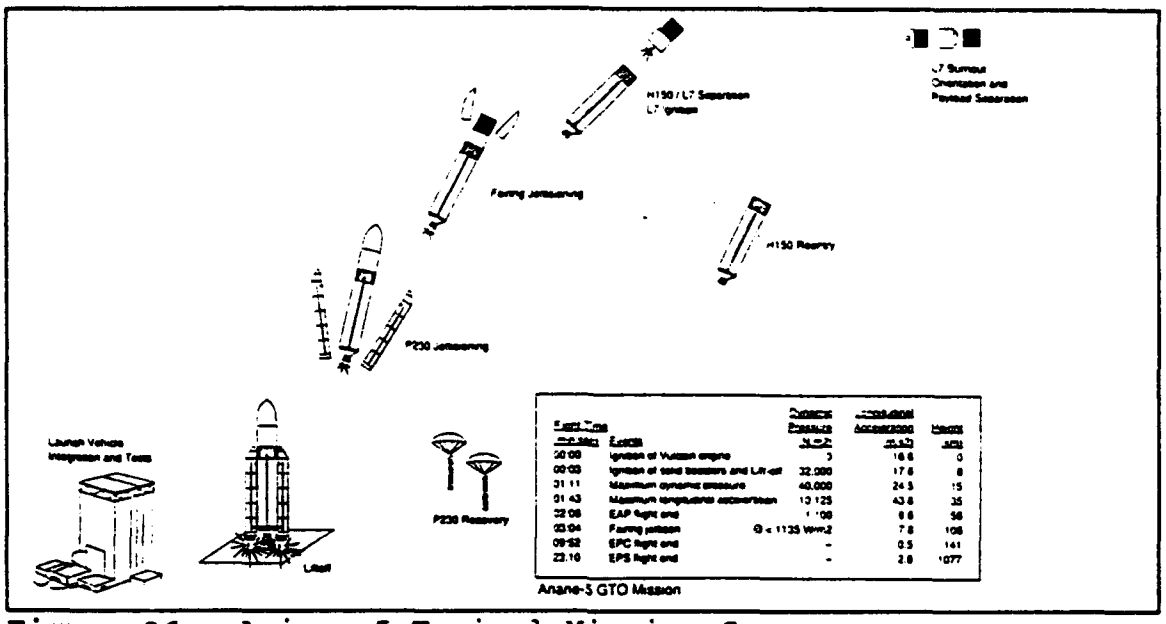

Figure 26. Ariane 5 Typical Mission Sequence. [Ref. 5:p. 44]

E. COST

The Ariane 5 is still being developed. Claude Quievre, Arianespace's vice president of technical affairs and production has said that the price for Ariane 5 services will be determined by the competitive situation in the market. He also said "Ariane 5 should have a lift capacity double that of our baseline competitor, the Atlas Centaur." He added, "So our sales price objective for an Ariane 5 should be twice that of the price offered for a commercial Atlas Centaur mission." [Ref. 21:p. 22]

First estimates for an Ariane 5 launch seem to indicate a price tag of approximately 100 - 110 million 1990 U.S.

dollars. [Ref. 5:p. 34] The first test launch of the Ariane 5 launcher is scheduled for 1995, with commercial operations to start in 1996. [Ref. 21:p. 22]

VI. LONG MARCH

A. HISTORY

The foundation of the Chinese space program can be traced back to the mid 1950's and China's Twelve Year Development Plan of Science and Technology. As a developing nation of the era, China recognized the need to develop a space capability. Research and development of a space launch program began in the 1960's. [Ref.5:p. 8]

The Chinese space launcher program began like a majority of the earliest space programs, as an offshoot of a ballistic missile program. During the initial stages of the Cold War, China relied heavily on obsolete Soviet missiles and locally produced versions. Initial missiles used by the Chinese included the SS-2, Sibling, and later the SS-3, Shyster. [Ref. 22:p. 103]

In the 1960's, after the break between Peking and Moscow, the Chinese developed its first indigenous Intermediate Range Ballistic Missile (IRBM), the CSS-1 (Chinese Surface to Surface Missile Number 1). The CSS-3, a follow-on to the CSS-1 and CSS-2, is the basis for the civilian version called Chang Zheng 1 (CZ-1), more commonly known in the West as Long March 1. [Ref. 22:p. 105]

1. Development

The first successful launch of a Chinese launch vehicle occurred on 24 April 1970 with the launch of a Long March 1 carrying a Dong Fang Hong 1 satellite. The CZ-1 was a three stage booster developed from the CSS-3. It was 99.9 feet tall and had an outside diameter of 7.38 feet. This launch vehicle was capable of placing 660 lb into LEO. [Ref. 8:p. 243]

The Long March 2 began its development in 1970. An interesting note is that the CSS-4, IRBM, was developed in parallel with this launcher. The CZ-2 is a two stage launcher designed to place 4,800 lb into LEO. The four first stage engines used N_2O_4 and UDMH as a fuel. The first stage attitude control was provided by gimbaling the engines while the second stage was controlled by using four verniers. [Ref. 5:p. 8]

The first launch of the CZ-2A ended in failure after only a few seconds. This was the only attempted flight of the A version. The CZ-2C, the follow-on to the A version, flew its first successful flight on 26 November 1975. The CZ-2C is capable of placing 7,040 lb of payload into LEO. The major difference between the 2C and the 2A was the improved reliability and performance of the launcher.

The next member of the Long March family, the Long March 3, began its development in 1977. This was a three stage vehicle which used the first two stages of the CZ-2C and

a cryogenic third stage. With the first launch of a CZ-3 ending in failure on 29 January 1984, a second launch was made only three months later. The successful flight of the CZ-3 on 8 April 1984 meant that the Chinese were the third country to use cryogenic technology at that time, the United States and France being the other two nations. [Ref. 21:p. 107]

The CZ-3 is 144 feet tall and has a diameter of 11 feet. It is capable of placing 11,000 lb into LEO, but more importantly, it was able to place 3,300 lb into a GTO. One of the enhancements made from the CZ-2C basic design was the incorporation of aerodynamic fins on the first stage.

The CZ-4 is based on a stretched version of the Long March 2C, and the third stage uses storable propellants. This launcher stands 138 feet tall and has a diameter of 11 feet. The payload capacity is 8,800 lb to LEO and 2,430 lb to a GTO. Originally designed to carry Chinese geostationary communications satellites, its lift capability was not great enough, and the vehicle has been employed launching polar orbiting Earth resource satellites. The first successful launch occurred 7 September 1988.

The current heavy lift vehicle, in the large array of Chinese launchers, is the Long March 2E. First launched in 1990, it is capable of placing 20,430 lb into LEO and 7,430 lb with a perigee kick motor (PKM) into a GTO. [Ref. 5:p. 8]

The LM-2E is composed of a stretched version of the LM-2C with the addition of four strap-on, liquid propellant

boosters. The total rocket stands 190 feet tall and has a core diameter of 11 feet. The primary commercial use of the LM-2E will be the placing of large spacecraft into a GTO. Figure 27 shows the CZ-2E.

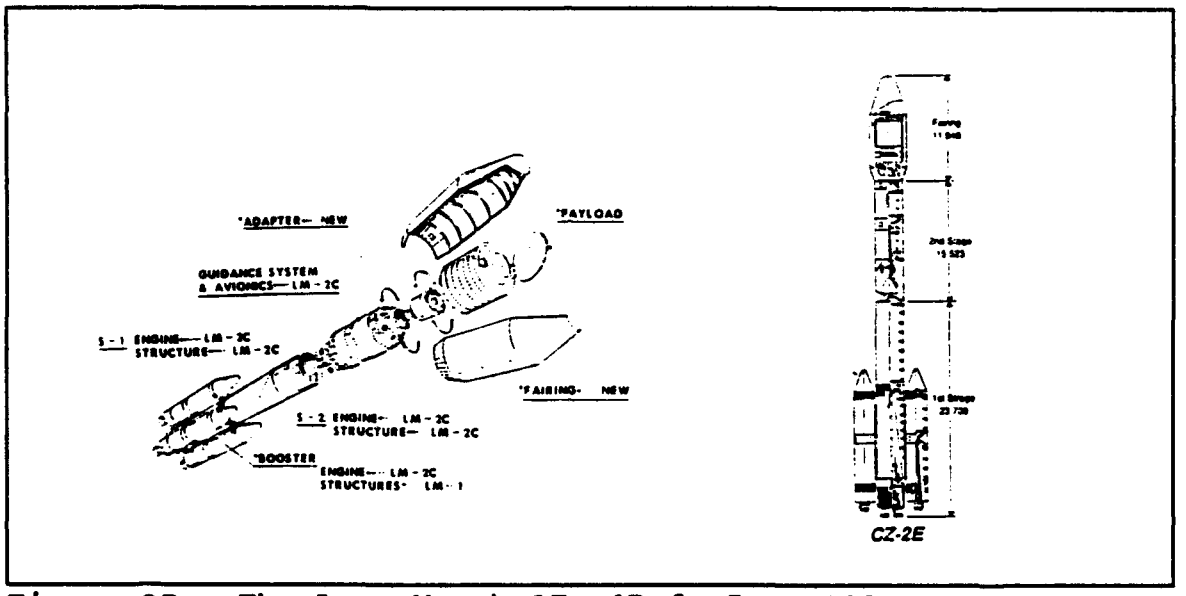

Figure 27. The Long March 2E. [Ref. 5:p. 13]

2. Success Rate

The overall success rate of the Chinese space launch program was 79.4% based on seven failures out of 34 launch attempts, through December of 1991. The Long March family success rate is 88.5% with 23 successful launches out of 26 attempts. The Long March family launch history is shown in Figure 28.

The Long March 1 series of launchers had only two launches, the first in 1970 and the second in 1971. Both launches were successful. The LM-1 has a 100% success rate.

It is currently being brought back into production after a 20 year absence. The new launcher is designated the LM-1D. [Ref 5:p. 10]

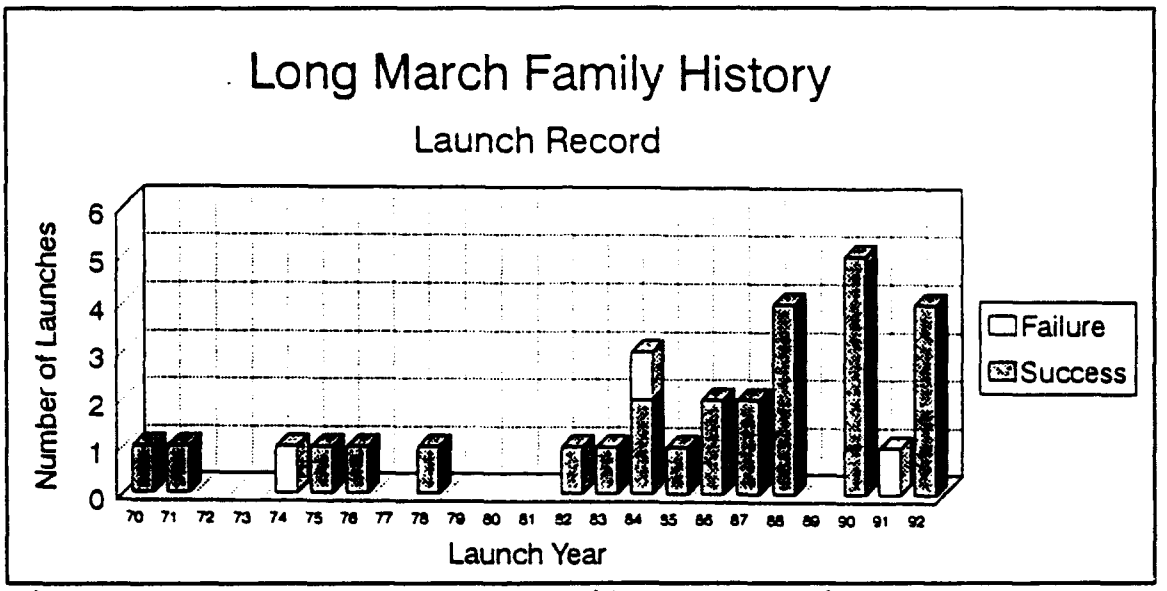

Figure 28. The Long March Family Launch History.

The Long March 2 family of launchers is still in production. The success rate for all the versions of the LM-2 models is 92.8% with only one failure in 14 attempts. The first failure occurred on the only launch of a CZ-2A model. The failure of this launch was due to a loss of attitude stability. The LM-2 family of launchers is the cornerstone for the Chinese commercial launch industry.

The Long March 3 launcher has had two failures in eight attempts. This gives the LM-3 a success rate of 75%. The first failure was attributed to not being able to restart the cryogenic third stage, and the payloads were both lost.

The final member of the Long March family, LM-4, is currently two for two in successful launches, giving it a 100% success rate.

Based on the successful performance of the Long March 2 launchers the decision was made to offer Chinese launchers commercially in 1985.

B. LONG MARCH 2E VEHICLE DESCRIPTION

The Long March 2E is a two stage launch vehicle with four strap on boosters, capable of placing 20,430 lb payload into LEO. It is based on the proven LM-2C design, and incorporates stretched versions of the first and second stages.

1. Strap on Boosters

The distinguishing feature of the CZ-2E is the use of strap-on boosters. This was the first Chinese Long March launch vehicle to use strap-ons.

Each strap-on booster (LB40) stands 52.5 feet high and has a diameter of 7.4 feet. The gross mass of each of the boosters is 90,000 lb of which the fuel makes up 84,000 lb. The LB40 is constructed of aluminum. [Ref. 5:p. 14]

A single YF-20 engine is used by each booster to provide 166,000 lb of thrust at sea level with an Isp of 289 seconds vac. The YF-20 is the same engine which is used in the first stage of the LM-2E. The engine burns a mixture of N_2O_4 and UDMH. [Ref. 8:p. 245]

After ignition the boosters will burn for approximately 128 seconds. These boosters are designed to burn to depletion with no throttling capability or restart capability. Booster separation is accomplished through the use of 16 retro rockets which initiate separation 1.5 seconds after booster burnout.

There is no real attitude control provided by the boosters. The engines are installed with a fixed cant to provide limited control. The bulk of attitude control is accomplished through the first stage.

2. Stage 1

The first stage of the Long March 2E, designated L180, is a stretched version of the first stage of the Long March 2C. It is 77.8 feet tall and has a diameter 11.0 feet. The original CZ-2C is 67.3 feet tall and has a diameter of 11.0 feet. The gross weight of the first stage is 433,000 lb with the fuel accounting for 412,000 lb. [Ref. 5:p. 14]

The first stage is composed of six sections, the interstage section, the oxidizer tank, the inter-tank section, the fuel tank, the aft skirt and a tail section. The tanks and inter-tank sections are made from an aluminum alloy. The aft transition section is made by welding together four chemically milled ribbed panels; this provides the mounting for the engines. The tail section is constructed from two half shell structures butt-jointed and riveted together.

The first stage uses four YF-20 engines. The specifications for these engines are the same specifications given for the engine used in the LB40. Figure 29 shows the cluster of YF-20 engines.

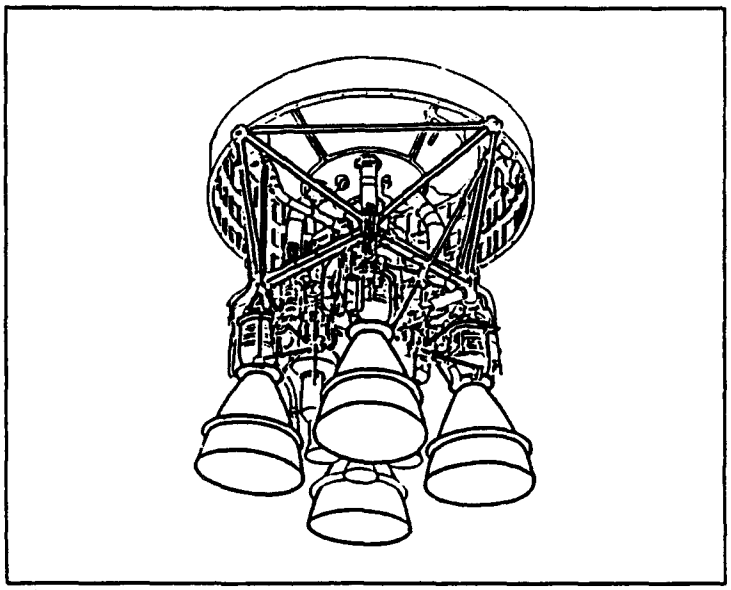

Figure 29. First Stage YF-20 Engines. [Ref. 5:p. 14]

The first stage has a nominal burn time of 159 seconds. There is no throttling capability and no restart capability for the first stage. This stage is also designed to burn to depletion.

Attitude control is provided by hydraulic gimbaling. This method of attitude control allows for pitch, roll and yaw control of $\pm 10^{\circ}$. Each engine can gimbal in one direction. [Ref. 5:p. 14]

Ideally the first stage burns for 158.9 seconds. The second stage would ignite at 159.7 seconds using the 'fire-in-hole' method. First stage separation occurs at 160.4 seconds into the mission. [Ref. 8:p. 245]

3. Stage 2

The second stage of the LM-2E, designated L90, is once again a stretched version of the second stage of the LM-2C. The L90 stands 50.9 feet tall, over twice as tall as the LM-2C second stage, and is 11.0 feet in diameter. The stage has a gross mass of 202,000 lb with the propellant having a mass of 190,000 lb. Figure 30 shows the second stage engine configuration.

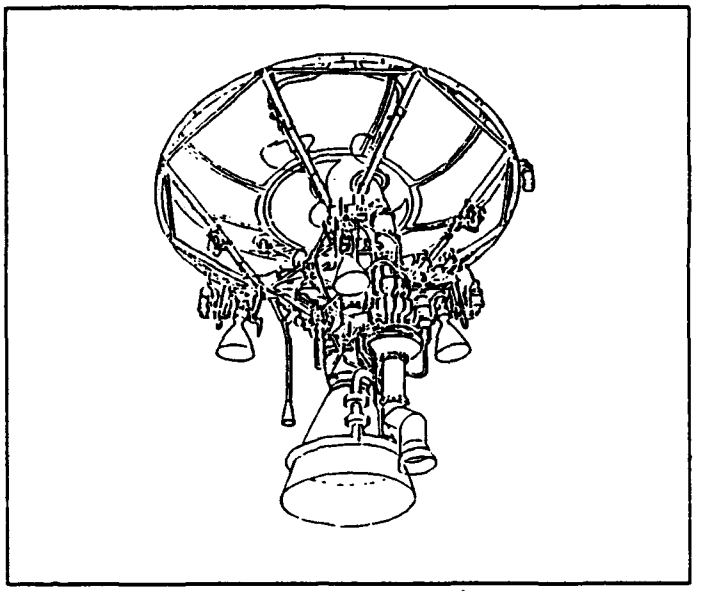

Figure 30. Stage 2 Engines.
[Ref. 5:p. 15]

The second stage uses five rocket motors. A single YF-22 is the main engine and four YF-23 engines are used as verniers. The YF-22 is very similar to the YF-20 used as the propulsion mechanism for the first stage and strap-on boosters. It is the high altitude variant of the YF-20. It provides 161,000 lb vac of thrust with an Isp of 296 seconds vac. It uses nitrogen tetroxide as the oxidizer and UDMH as the fuel, the same propellant combination as the first stage. The nominal burn time for the second stage is 295 seconds. The main engine has no restart capability and no throttling capability. [Ref. 8:p. 248]

The four YF-23 engines used by the second stage are gimbaled and are used to provide steering control for the YF-22 of the second stage. The four engines provide a total of 9,900 lb vac thrust with a specific impulse of 218.7 seconds vac. The engines use the same fuel as the YF-22 and carry enough fuel for a total burn time of 410 seconds. [Ref. 8:p. 248]

Attitude control for the second stage, as mentioned above, is accomplished through the use of a gimbaled YF-22 and the four YF-23 engines. Pitch, roll and yaw can be controlled within $\pm 60^{\circ}$ through the use of the five engines. [Ref. 5:p. 15]

C. PAYLOAD SECTION

At the present time there is one standard fairing being used with the Long March 2E. This fairing is 39.2 feet tall and has an outside diameter of 13.8 feet. (See Figure 31)

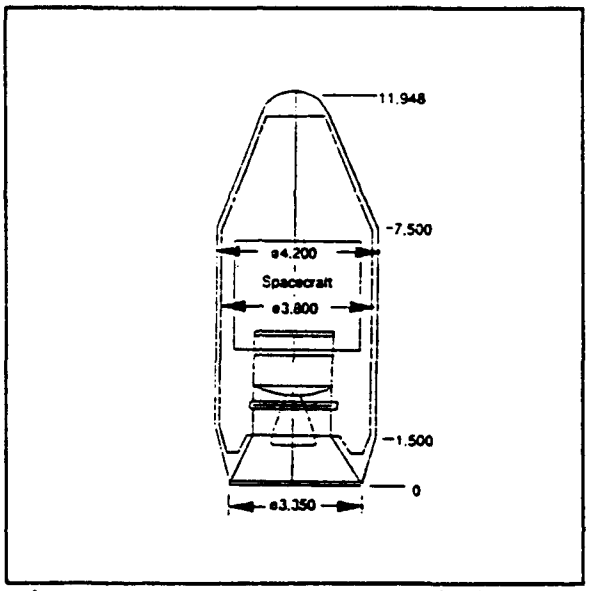

Figure 31. Payload Fairing. [Ref. 5:p. 17]

The maximum payload which can be housed in the fairing has a diameter of 149.6 inches, a maximum cylinder length of 236.2 inches and a cone length of 135.2 inches. There are currently three adapters for integrating payloads and these adapters have diameters of 47.1 inches, 63.1 inches, and 64.8 inches.

The fairing is designed as two half shells using longitudinal separation. Phenolic resin glass cloth is used to manufacture the nose dome; the remainder of the fairing is made of an aluminum honeycomb.

Payload access is provided through eight access doors built into the fairing. Radio transparent windows are also built into the fairing and maybe opened at the user's request; radio transparency is advertised as not less than 85%. With the launcher on the service tower, the payload compartment may be air conditioned, with the temperature controlled between 41-59° Fahrenheit with a relative humidity not greater than 55%. There is audio insulation mounted to the innerwall of the fairing to help minimize the noise affecting the payload during flight. [Ref. 5:p. 17]

D. TYPICAL MISSION PROFILE

Currently the Long March 2E is only being launched from the Xichang Satellite Launch Center (XSLC). The Jiuquan Satellite Launch Center (JSCL) and the Taiyuan Satellite Launch Center (TSLC) are being adopted for use by the entire range of Long March launchers. Figure 32 shows a typical mission sequence.

The time required to place a CZ-2E in orbit from the time it is erected on the launch pad to liftoff is approximately 11 days. This is reduced from the CZ-3 ideal time requirement of 20 days from the first stage being stacked until launch. [Ref. 23:p. 185]

At T-11 days from launch, the LM-2E is erected and assembled. Two days are required to perform this function. During this time checkout preparation on the stages is also

accomplished. Once the stages are assembled, the entire launch vehicle undergoes another checkout procedure; this one lasts approximately two days. Seven days prior to launch the payload is integrated and the fairing assembly is installed on the launcher. Six days prior to launch a complete checkout of the entire launcher and payload is accomplished. Four days prior to launch a prelaunch rehearsal is held.

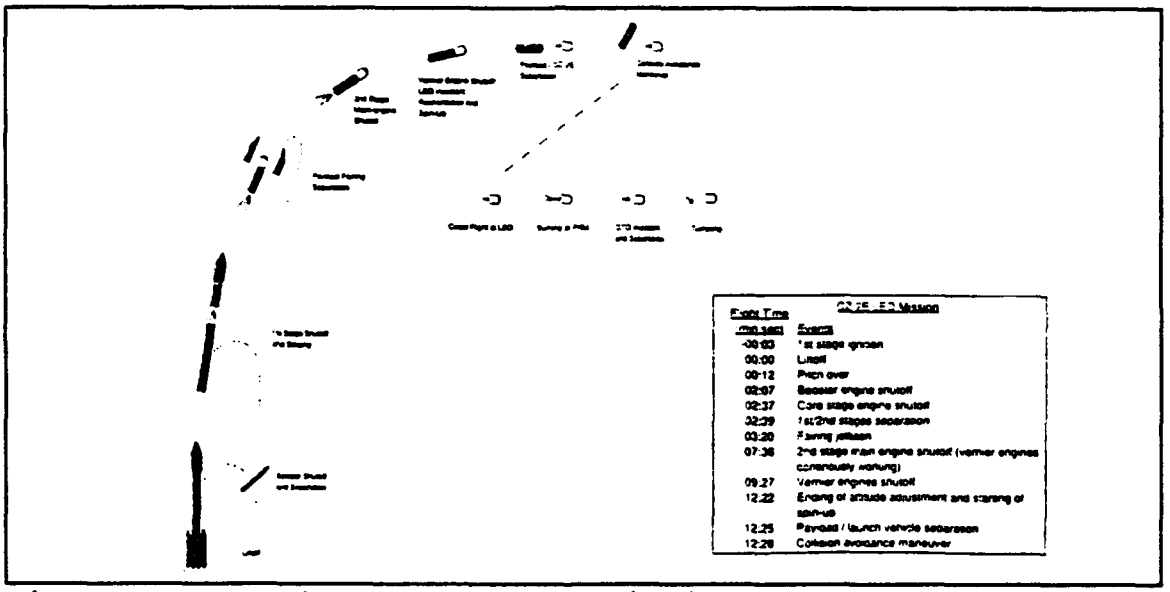

Figure 32. Typical Long March Mission. [Ref. 5:p. 24]

Five hours before launch the vehicle undergoes vertical adjustment and aiming. At T-4 hours the system undergoes charging and prelaunch functional checks. Two hours prior to launch the work platform is moved back. At T-40 minutes, pressurizing lines and fueling lines are removed. The switch to onboard telemetry occurs at T-20 minutes. At T-1 minute

the swing cable rod is withdrawn. At T-03 seconds the first stage engine is ignited. [Ref. 5:p. 23]

Liftoff occurs at 00:00 (minutes:seconds). At 00:12 a pitch-over maneuver occurs to place the vehicle in the proper trajectory for the desired mission. The strap on booster engines shut off at 02:08, with booster separation occurring 1.5 seconds later at time 02:09. The first stage burns for 160 seconds with shutdown occurring at time 02:37. The second stage engine ignites at time 02:38 and stage separation occurs at time 02:39. The payload fairing is jettisoned at 03:20. The second stage main engine burns until 07:36 at which time it shuts down. The verniers have been firing continuously during this time. The verniers shut off at time 09:27. Satellite spin-up occurs at the completion of the attitude adjustment phase which happens at the 12:22 point of the mission. Payload separation for a LEO mission occurs at time 12:25. [Ref. 5:p. 24]

E. COST

The China Great Wall Industry Corporation (CGWIC) is the foreign trade company responsible under the Ministry of Astronautics for marketing and negotiating launch services. CGWIC has many functions in the import and export of Chinese astronautics technology and products. Launch services is one part of the CGWIC. In 1990 the price of 30 million U.S. dollars was quoted as the price for a Long March 2E LEO

mission. [Ref. 8:p. 245] Other sources have set the price of a CZ-2E mission at around 40 million U.S. dollars (1990).

Since 1985, when the Chinese first announced willingness to provide commercial launch services, U.S. and European competitors have criticized China for offering launch services at unfairly low prices. Yu Xianrong, assistant to the president of Great Wall Industry Corp. stated, "To attract users it makes sense to offer a discount." Yu went on to say, "But we are a commercial organization. Our launch vehicle manufacturer calculates their cost and we discuss the price. Each time we have a thorough discussion." [Ref. 24:p. 28]

The concern over low cost launches eventually produced an agreement in December of 1988 that the Long March launchers would be limited to a total of nine international satellite launches through 1994. The cost of these launches should be compatible with the international launcher market. [Ref. 8:p. 242]

VII. H-2

A. HISTORY

The Japanese government became interested in space flight in 1964 when satellite images of the Tokyo Olympics were broadcast to the United States. Prior to that time only some limited work was done using sounding rockets. In 1964 the Japanese Science and Technology Agency created the National Space Development Center to determine if any practical benefits could be obtained through the use of space. The center then became the National Space and Development Agency of Japan (NASDA) in 1969.

In 1969, the United States and Japan concluded an agreement that provided for the transfer of Delta launch vehicle technology. This transfer of technology led to the development of the N-1 launch vehicle. The McDonnell Douglas Corporation, the maker of Delta, provided much of the overall design, production and launch operations for the N-1 launcher. The Japanese government, in return for all the technological support, was prohibited from offering the N-1 commercially. The N-1 led to the H-1 which has led to the all-Japanese H-2.

1. Development

The Japanese H-2 program can trace its beginnings to the N series of rockets. The first successful N-1 launch vehicle was launched on 9 September 1975 from the Osaki Launch Site (OLS) at the Tanegashima Space Center.

The N-1, was a three stage launcher, which utilized three solid propellant Castor strap-on boosters, capable of delivering 290 lb into a geostationary orbit. It was derived primarily from the United States Delta launch vehicle. The N-1 stood 107 feet tall with a core diameter of eight feet. At liftoff the vehicle had a total weight of 199,000 lb. [Ref. 5:p. 72]

The N-1 was flown from 1975-1982. During this time period, due to increased technology and an increased demand, communications satellites became larger and heavier. The N-1 was not capable of placing the newer commercial communications systems in orbit, so a heavier lift vehicle was needed.

The N-2 was designed to replace the N-1. The N-2 made its first successful launch on 11 February 1981 from OLS. The N-2 was based on the proven N-1 design and incorporated nine solid propellant Castor II strap-on boosters. Other improvements over the N-1 included an extended first stage tank, improved second stage engine performance and an improved inertial guidance system.

The N-2 was 116 feet tall and eight feet in diameter. It had a total weight of 297,000 lb and was capable of placing

770 lb of payload into a Geostationary orbit, or 4,400 lb into LEO. Eight N-2 vehicles were launched between 1981 and 1987. Once again the payload requirements for the new generation of satellites was too much for the current Japanese launcher. Development had been in progress since 1977 for a replacement system for the N series of launchers; this was to become the H series of launch vehicles. [Ref. 5:p. 73]

The H-1 launch vehicle utilized the first stage from the N-2 and the nine strap-on boosters. The second stage was a cryogenic stage, utilizing liquid oxygen and liquid hydrogen, which had been designed and built in Japan. Other improvements included a domestically designed inertial navigation system, and the third stage solid rocket motor.

The first H-1 was launched on 13 August 1986 from OLS. The H-1 launch vehicle is 132 feet tall and has a core diameter of eight feet. The total weight at liftoff of the H-1 was 308,000 lb, and it was capable of placing 1,200 lb in GEO and 7,000 lb in LEO. [Ref. 5:p. 72]

Even before the first launch of the H-1, development had begun on the all Japanese H-2 launcher. Development began in 1985 and the vehicle base line configuration was established at the Preliminary Design Review in May of 1987. [Ref. 25:p. 378]

The H-2 is 161 feet tall and has a core diameter of 13.1 feet. It has a total weight of 582,000 lb and is capable of placing 4,800 lb in GEO or 23,000 lb in LEO. The H-2 is a

two stage vehicle augmented by two solid propellant rocket motors. The first and second stages are both cryogenic stages. [Ref. 25:p. 378] Figure 33 shows the H-2 launcher.

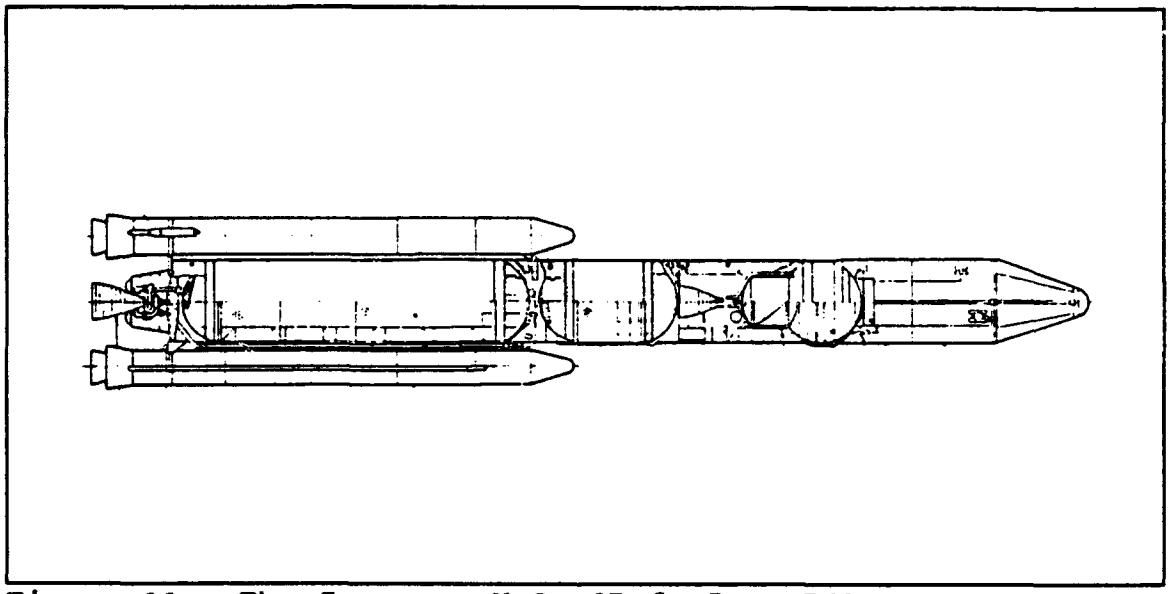

Figure 33. The Japanese H-2. [Ref. 5:p. 76]

The first successful launch of the H-2 occurred on 4 February 1994 and marks a major step for the Japanese space program. Mastato Yamano, Chief of NASDA, one hour after the launch stated, "Our catch-up period is over. We want to lead the world." [Ref. 26:p. 1]

2. Success Rate

The Japanese N series and H series of launch vehicles has been extremely successful. Not a single failure has been reported during any of the 25 launches through February of 1994. The N series and H series launcher history is shown in Figure 34.

The N-1 launcher had a 100% success rate. The last launch of an N-1 was on 3 September 1982. That marked the seventh successful launch of an N-1 launcher.

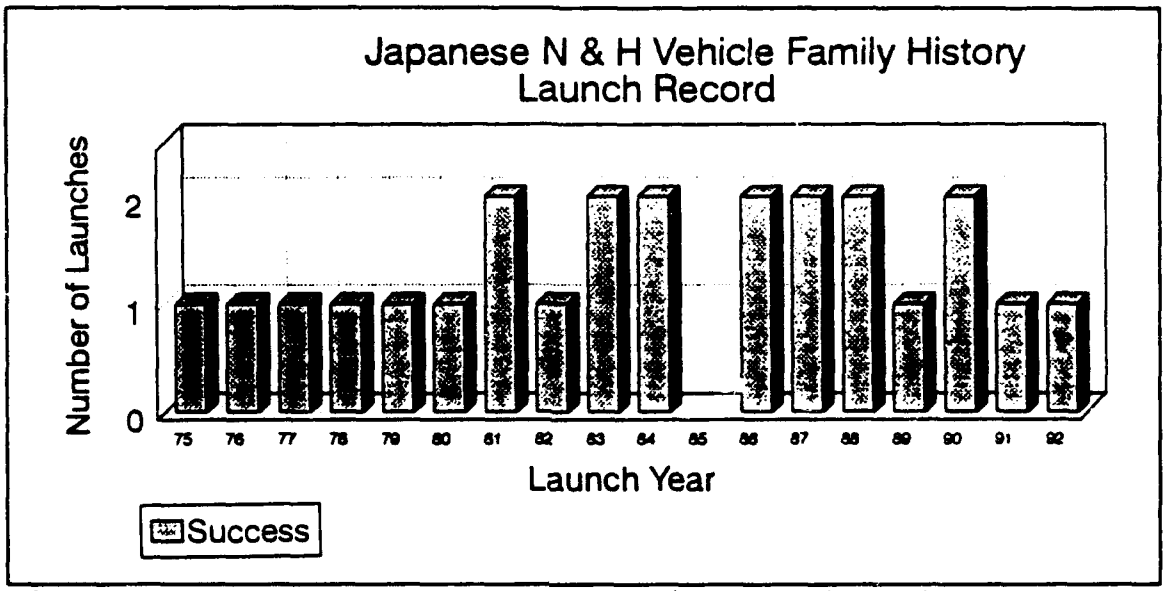

Figure 34. The N and H Launch Vehicle Family History.

The N-2 has been just as successful. It maintained a flawless launch record through eight launches. The final launch of an N-2 occurred on 19 February 1987.

The H-1 launch vehicle also has an unblemished launch record. Since its first launch in 1987, the H-1 has been used successfully in nine consecutive launches.

The future workhorse for NASDA, the H-2, made its successful maiden voyage on 4 February 1994.

The unprecedented success rate of the Japanese launcher program may be a deciding factor when it comes to selecting a launch vehicle option in the future.

B. H-2 VEHICLE DESCRIPTION

The H-2 is the first launcher to be developed by NASDA without any foreign technology. The chief of NASDA, Masato Yamano, noted "... and we are very proud of this fact," when talking about the all Japanese development of the H-2. [Ref. 26:p. 1]

The H-2 consists of two stages and two solid rocket boosters (SRBs). The prime contractor for the H-2 is Mitsubishi; it leads a consortium consisting of Nissian Motor Company, IHI, NEC, Kawasaki Heavy Industries, Fujitsu Ltd, Japan Aviation Electronics Industry and Toshiba Corp. The consortium is known as the Rocket Systems Corporation (RSC). [Ref. 8:p. 285]

1. Solid Rocket Booster (SRB)

Two SRBs are used on the H-2. Each SRB is 76.8 feet tall with a diameter of 5.94 feet. Each booster weighs 155,000 lb of which the solid propellant makes up 131,000 lb. Each booster is composed of an aft skirt, forward adaptor, nose cone and four solid rocket motor segments. [Ref. 5:p. 78] Figure 35 shows the H-2's SRB.

The aft skirt houses separation motors and the thrust vector control (TVC) system. The forward adaptor houses additional separation motors and the majority of the booster's electronic components. [Ref. 5:p. 78]

The strap-on engine is designed by the Nissan Motor Company, Ltd. It is Japan's largest and most powerful indigenously developed solid rocket motor. It is composed of four segments, with the aft three segments having a center port design and the forward segment having a five point star grain configuration. Each segment is composed of 14% hydroxy terminated polybutadiene (HTPB), 18% aluminum and 68% ammonium-perchlorate. The case for each segment is rolled from a low alloy carbon steel. The segments are joined by a bolted flange which consists of one O-ring and 108 bolts. [Ref. 5:p. 78]

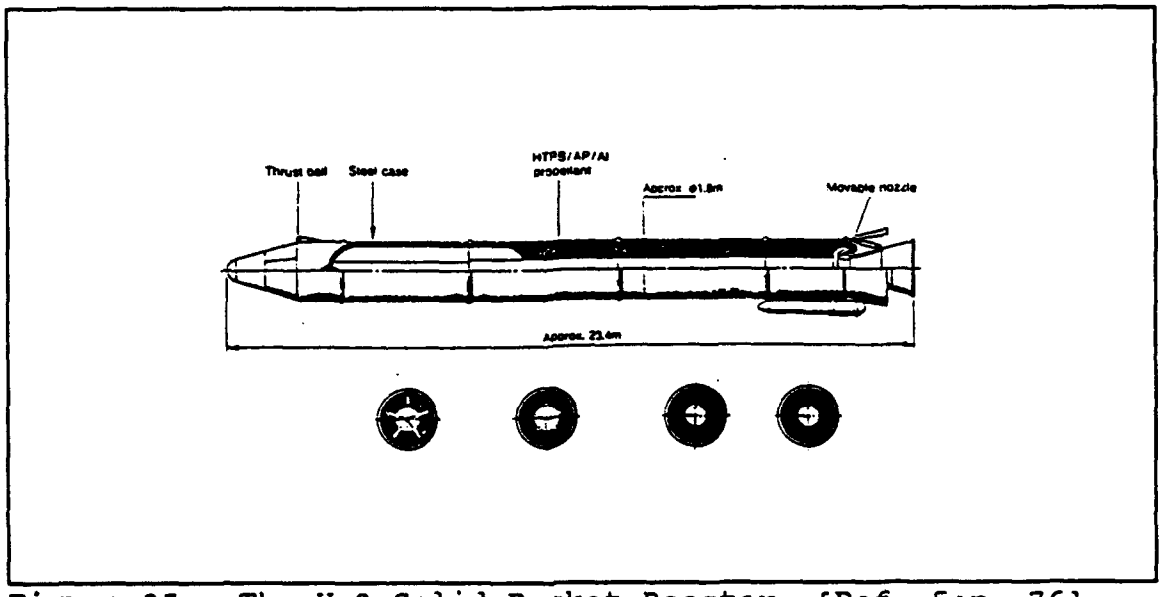

Figure 35. The H-2 Solid Rocket Booster. [Ref. 5:p. 76]

Each SRB is designed to produce 351,00 lb of thrust at sea level with an Isp of 273 seconds vac. It is designed as a burn-to-depletion motor with no restart capability. The nominal burn time for the SRB is 94 seconds. [Ref. 8:p. 287]

Attitude control is provided through the TVC. TVC is accomplished through a movable nozzle with an aft pivoted flexible joint that provides omniaxial deflection capability of 5 degrees during motor burn. [Ref. 25:p. 379]

2. Stage 1

The first stage of the H-2 represents the largest technological design increase over the H-1. It is 95 feet tall and has a diameter of 13.1 feet. Of the 216,000 lb that make up the mass of the first stage, 190,000 lb are from the fuel. The heart of the first stage is the Mitsubishi Heavy Industries Limited, LE-7 engine. [Ref. 5:p.78] Figure 36 shows the first stage.

The LE-7 is Japan's largest and most advanced engine. It is a high pressure, cryogenic, staged-combustion[6] cycle engine. The engine is designed to produce 190 lb thrust at sea level with a specific impulse of 445 seconds vac. It uses liquid oxygen and liquid hydrogen. [Ref. 8:p. 293]

Fuel and oxidizer are stored in tanks in the first stage. Each tank consists of two spherical bulkheads and an isogrid-processed cylinder. The tanks are made out of 2219 aluminum and have sprayed on polyisocyanurate foam as an insulation. The tanks contain anti-slosh baffles, structural

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[6]Staged combustion occurs when the propellants are partially burned in the pre-burner, routed to drive the turbopumps and then combined with more oxygen in the main combustion chamber.

support frames, level sensors and temperature sensors. [Ref. 5:p.78]

The first stage is designed to burn for 348 seconds. There is no throttling control and no restart capability. Pitch and yaw control for the first stage are provided through hydraulic gimbaling of the main engine. The LE-7 is designed to allow for ±7° of control. Additionally, auxiliary engines for attitude control in all three axis are used on the first stage during powered flight and the coast phase. These engines use hydrogen gas from the main engine and cold nitrogen gas, stored in a nitrogen bottle. [Ref. 5:p.78]

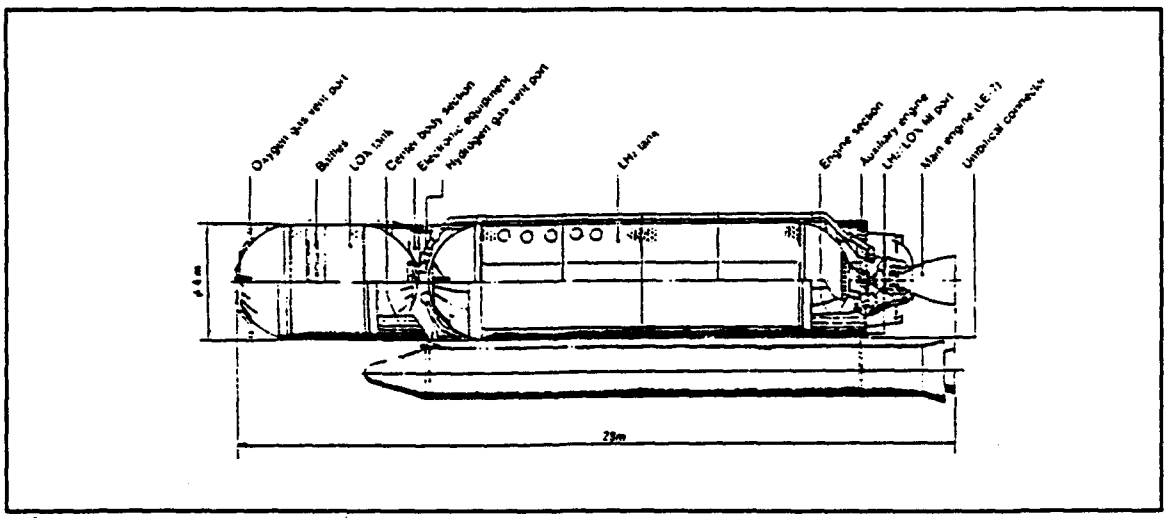

Figure 36. The First Stage of the H-2. [Ref. 5:p. 37]

3. Stage 2

The second stage of the H-2 is a modified version of the second stage of the H-1. The second stage is 35.8 feet tall and has a diameter of 13.1 feet. It has a gross mass of

43,400 lb of which the fuel accounts for 36,800 lb. This is about a 60% increase in propellant by weight. The second stage is shown in Figure 37.

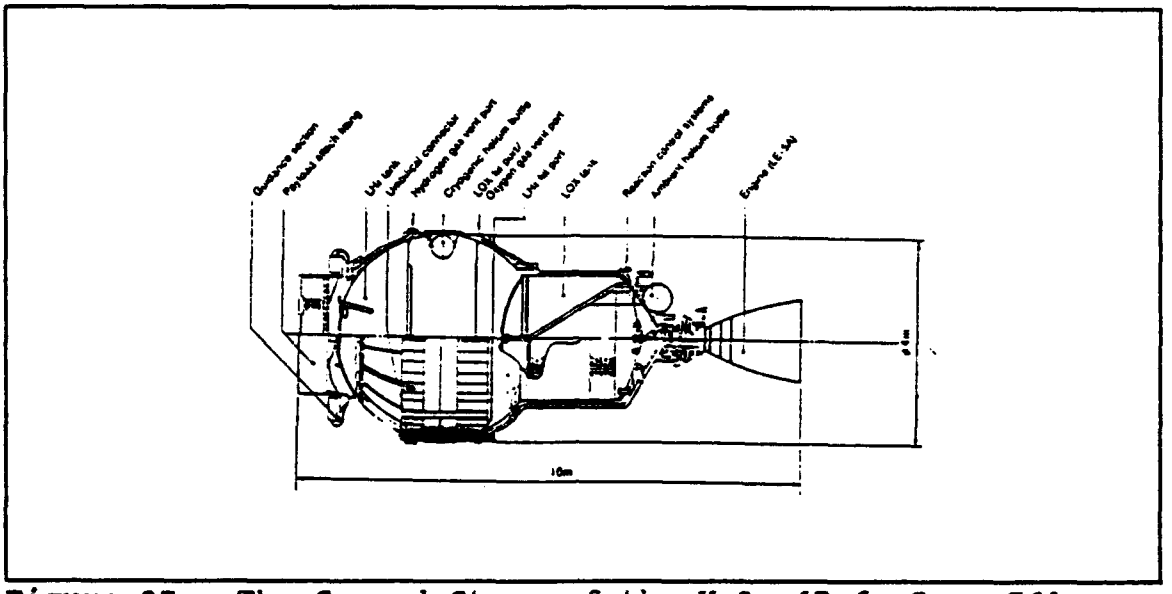

Figure 37. The Second Stage of the H-2. [Ref. 5:p. 76]

An upgraded version of the H-1's LE-5 engine is used, designated the LE-5A. The changes include uprated performance and improved restart capability. This is done by increasing the chamber pressure and throat diameter and by switching from gas generator to a hydrogen bleed cycle[7]. The use of the hydrogen bleed cycle eliminates the need for a gas generator and therefore simplifies design. The LE-5A engine produces

[7]The hydrogen bleed cycle operates by having the turbopumps driven by gaseous hydrogen from the nozzle cooling jacket.

27,300 lb vac thrust with an Isp of 452 seconds vac. [Ref. 5:p. 78]

The LE-5A engine is capable of multiple restarts. It is designed as a command shutdown stage and some throttling is available.

The tanks of the second stage are also made from 2219 aluminum alloy and constructed using an isogrid structure. The increased tank size from the H-1 second stage means that more fuel and a longer burn can be had. The second stage is designed to burn for 590 seconds which is an increase of 220 seconds over the H-1 second stage. [Ref. 5:p. 78]

Attitude control for the second stage is provided through the use of a hydraulically gimbaled engine. The gimbaling allows for $\pm 3.5^{\circ}$ of correction in the pitch and roll axis. In addition a reaction control system (RCS) is used for yaw correction and pitch and roll correction during coast phase. The guidance control package is located at the top of the second stage. [Ref. 5:p. 78]

C. PAYLOAD SECTION

There are currently three different fairings which can be used with the H-2. The standard fairing has an outside diameter of 160.2 inches and an overall length of 472.4 inches. A longer, 590.5 inches, version is also available for dual payloads. The third variant is the large 196.9 inch

diameter version. It is also about 590.5 inches long. [Ref. 25:p. 379] Figure 38 shows the payload section options.

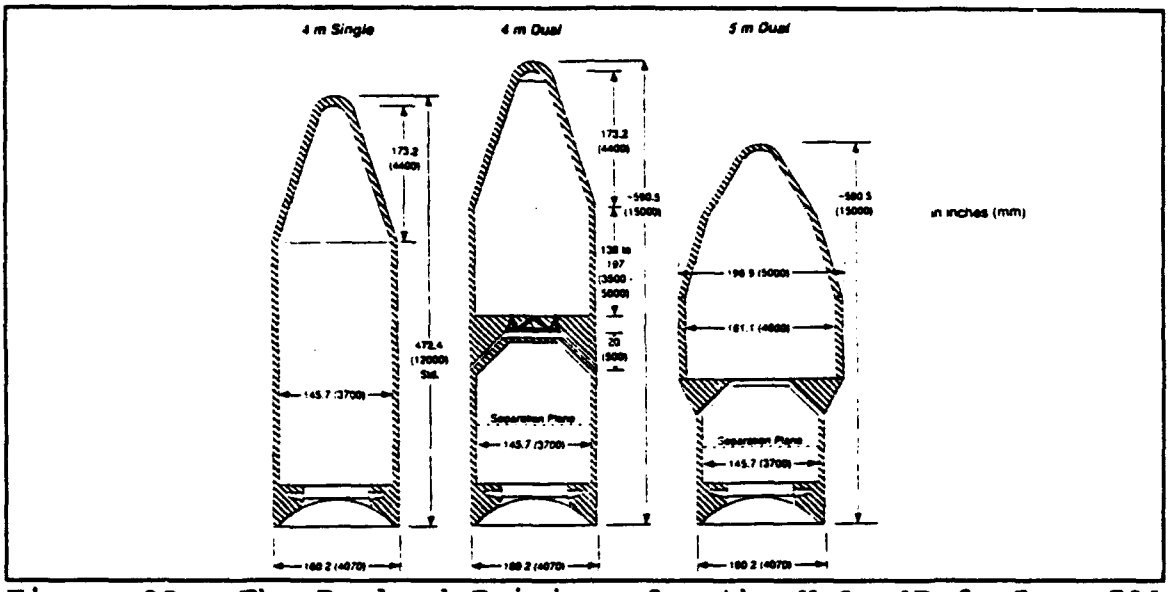

Figure 38. The Payload Fairings for the H-2. [Ref. 5:p. 79]

All the fairings are composed of an aluminum skin over an aluminum honeycomb core. The surface is coated with composite material, including silica micro-balloons, which provides thermal insulation. Acoustic attenuation is provided by the use of acoustical absorbing blankets which are located on the inside of the fairing. [Ref. 5:p. 79]

The fairings are constructed from two half-shell pieces. A frangible bolt type longitudinal separation system is used. A spring separation system is also used to ensure separation. Gas pressure generated by an explosive cord fractures the frangible bolt for separation. This marks the first time that such a large, lightweight fairing has been developed in Japan. [Ref. 5:p. 79]

The maximum payload diameter for the smaller fairing is 145.7 inches, while for the larger diameter fairing the maximum diameter allowed is 181.1 inches. The maximum cylinder length is 137.8 to 196.9 inches for a dual payload. Access to the payload is usually up to T-10 hours through access doors in the fairing. [Ref. 5:p. 86]

D. TYPICAL MISSION PROFILE

The normal mission schedule begins 24 months prior to launch. The launcher is constructed at the site in a Vehicle Assembly Building (VAB). A typical mission profile is shown in Figure 39.

The SRBs are assembled horizontally in the VAB and then hoisted on to the Mobile Launcher (ML). The SRBs are then secured to the ML by four explosive bolts. The core stages are then assembled and attached to the SRBs. The entire weight of the vehicle is supported by the SRBs while on the ML. Once assembled and all checks completed, the ML travels 1640 ft to the launch pad. [Ref. 5:p. 84]

The Pad Service Tower (PST) consists of a 246 foot fixed tower with umbilicals which reach to the H-2. Two rotating towers are also located there. One tower is for access to the H-2's payload and the other is for access to the vehicle itself. The payload is mated to the fairing in the Fairing and Satellite Assembly Building. The payload, once enclosed

in the fairing, is hoisted atop the second stage and prepared for launch. [Ref. 5.p. 84]

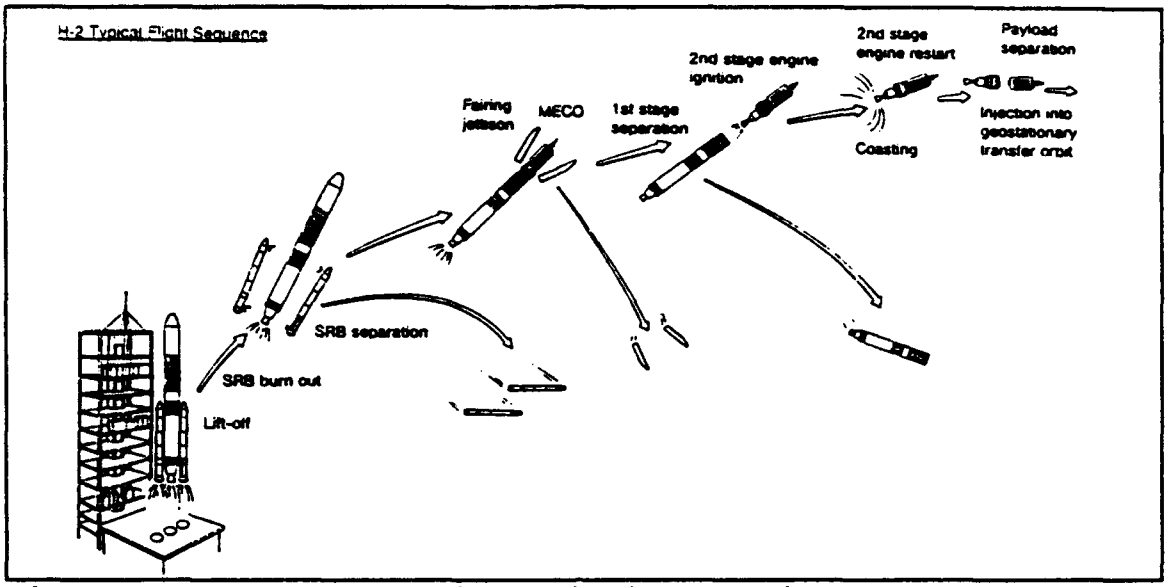

Figure 39. The H-2 Typical Mission Profile. [Ref. 5:p. 85]

At T-00:06 (minutes:seconds) the first stage engine ignites. Ignition of the SRBs occur at time 00:00, and liftoff occurs. At 01:37 the solid rocket boosters have burned out and five seconds later they are jettisoned. The payload fairing is jettisoned at time 04:40. The first stage continues to burn until 05:20 at which time it shuts down, and then eight seconds later stage separation occurs. The second stage ignites and burns until 10:46 when it is commanded to stop. The second stage is re-ignited at 23:45 and burns for an additional 200 seconds and shuts down at 27:05. Payload separation occurs at time 27:25, which signifies mission completion. [Ref. 5:p. 85]

E. COST

The H-2 places Japan in an ideal position to enter into the commercial launcher market. Its strong historical performance could make it a strong contender with Arianespace and Titan.

Masato Yamano commented after the successful H-2 launch in February of 1994 that "H-2 is expensive - we need to get the cost down." He went on to say "We will try to come up with a cheaper rocket. The number of parts may be cut down, or we may buy parts from abroad." The cost of the February launch was put at 150 million U.S. dollars. [Ref. 26:p. 20]

Estimates on the price of a launch of an H-2 have been put at about 100-120 million 1990 U.S. dollars. [Ref. 5:p. 74]

VIII. CONCLUSIONS

Putting heavy payloads into space is not an impossible mission. The launch vehicles described earlier all have the capability of launching over ten tons of payload into a low earth orbit. All of the vehicles, except the Ariane 5, have been successfully launched.

While many of the world's leading space powers have been investing money into new heavy launchers, the United States seems satisfied to keep using its tried and true Titan family of launchers. Japan, a relative newcomer to the space world, has invested a significant amount of money into its new H-2 rocket. Arianespace, the largest launcher of commercial vehicles, has invested and continues to invest in the Ariane 5 launcher. Do these companies and countries know something the United States does not?

Arianespace officials believe that the commercial satellite market is turning toward heavier satellites, which means that heavier lift capacities will be needed to launch the new generation of satellites. Estimates made by Arianespace in February of 1994 indicate that by the end of the decade more than 60 percent of all the commercial satellites will weigh between 2.4 tons and 3.6 tons. They also predict that 20 percent of the satellites will weigh over 3.6 tons. Charles Bigot, Arianespace Chairman, noted in

January 1994 that the Ariane 4 is becoming less and less profitable. This is because single heavy payloads are being contracted for, while before, dual payload launches were commonplace. "Ariane 5 is arriving at just the right time," he said. "Dual launches will enable us to restore profitability." [Ref. 27:p. 6]

The United States has not been blind to the changes in satellite requirements. The National Research Council (NRC) at the request of NASA conducted a requirements, benefits, technological feasibility and roles of Earth-to-orbit transportation options review. One of the conclusions reached was "The United States must make a long term commitment to new infrastructure and launch vehicles." Another recommendation of the NRC was "The 20,000-pound payload class, National Launch System (NLS-3) vehicle, should be the first of the proposed NLS family to be designed and built in coordination with the new launch facilities." [Ref. 2:p. 3]

The key element now for commercial launches is reducing costs. The managers of the Japanese H-2 and of the Arianespace Ariane 5 are examining ways to reduce production costs and thus make their launch vehicles more attractive commercially.

The need for heavy lift launchers is already here, but the question remains: is there a sufficient number of these heavier payloads to justify the expense of designing and

producing a new launch vehicle? Many nations have already answered this question with a yes.

The current budget reduction frenzy that has gripped the nation casts a dark shadow over any real chance of developing a new launch vehicle in the near future for the United States. On 1 September 1993, Secretary of Defense Les Aspin rejected an Air Force Proposal to develop a new space launch system, dubbed the Spacelifter. Eugene Sevin, the Director of Missile and Space Systems at the Pentagon, noted as a result of Aspin's decision the Air Force may have to make a more aggressive investment to keep its existing rockets in working order. He went on to say "Something more uncertain is how much we can improve the existing fleet and reach spacelifter goals without evolving a new vehicle." He warned against "Trying to build a Volkswagen into a Cadillac by part replacement." [Ref. 28:p. 4]

How lucrative is the commercial launch industry? According to an Arianespace market study, the 'open' commercial launch market from 1989 to 2000 is valued at $35-37 billion. A Florida congressman noted "The sale of one commercial launch by a US company is equivalent to the import of 10,000 Toyotas." [Ref. 29:p. 19]

One crucial aspect of Heavy Lift Launchers which was not covered is the political aspect of various launchers. This is a very unstable area. Ten years ago the idea of launching an American-built satellite on a Russian-built launcher was

laughable; today it is a reality. The idea of a "technology transfer" while still a concern is no longer a driving force behind international launches of United States satellites. Tim Furnish, a spaceflight correspondent for Flight International, noted "If the truth be known, the objection of the US is more to ensure that its fledgling ELV companies do not lose business - and face - to the Soviets." [Ref. 30:p. 247]

The China Great Wall Industry Corp. of Bejing as recently as March of 1994 has been attracting much attention by completing negotiations for three launches with options for 15 more in a three week time frame. Frank Weaver, director of the U.S. Department of Transportation's Office of Commercial Space Transportation, stated "When we hear reports of ... bids [from non-market economies] reflecting inordinately low prices or large numbers of launches, we have concerns regarding their compliance with the pricing and quantity provisions in [the trade] agreements." The Chinese company is supposed to be pricing satellite launch services on a par with U.S. and European launchers under a 1988 trade agreement. That agreement expires at the end of this year. [Ref. 31:p. 1]

With a multitude of launchers available, and the capabilities of these launchers all being comparable, cost and availability may be the deciding factors for launcher selection. The current backlog for launch services for the various launchers are 36 satellites for the Ariane family of

launchers, 6 satellites for the H-2, 46 satellites for the Proton, 41 launches for the Titan IV and only 4 satellites for the Long March 2E. [Ref. 27:p. 6] Emery Wilson, spokesman for Hughes Communications International Inc., may have best summed up industry's present outlook on launcher choice when he stated "The Long March is less expensive than other launcher vehicles that are available." He went on to say "It's a Great Wall decision on how they price launch vehicles and how that relates to the World Market." Wilson also so noted that Hughes is primarily concerned with securing reservations with different launch agencies to provide customers with the soonest possible launches. [Ref. 31:p. 29]

The number of commercially available heavy lift launch vehicles is currently high enough to meet the projected need. The question remains as to how each company and nation elects to market those launchers. In a competitive market when there is a greater supply than demand the price of one supplier's product must go down in order to maintain a market share. The Chinese currently have the ability to offer lower prices due to lower production costs. If the United States wants to stay in the market, either the manufacturers of launchers will have to retool and learn to cut production costs, or the government will have to get involved more than it already is and impose trade sanctions on non-participating countries to ensure a share for U.S. companies in the lucrative launcher market.

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